NASA CR-132495

PROGRAM FOR ESTABLISHING LONG-TIME FLIGHT SERVICE PERFORMANCE OF COMPOSITE MATERIALS IN THE CENTER WING STRUCTURE OF C-130 AIRCRAFT

PHASE III - FABRICATION

September 1974

by W. E. Harvill and A. O. Kays

Prepared under Contract No. NAS 1-11100 by
LOCKHEED-GEORGIA COMPANY
Marietta, Georgia

for

Langley Research Center
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

This Phase III-Final Technical Report is submitted in fulfillment of the requirements of Contract NAS1-11100 and reports contract effort from February 1973 through June 1974. Phase III consisted of the fabrication of three C-130H center wing boxes, selectively reinforced with boron-epoxy composites. Prior to initiating the fabrication phase of program activities, extensive advanced development work was conducted in Phase I, and a detailed design, including the necessary analytical and component test substantiation of the selected design, was conducted in Phase II. The Phase I and Phase II program activities were previously reported in NASA CR-112126 and NASA CR-112272. Subsequent program phases include ground/flight acceptance tests of the three reinforced center wing boxes. One of the wing boxes will be subjected to a complete static and fatigue test evaluation. Two of the wing boxes will be flown on C-130H aircraft for a period of three years to demonstrate the long-time capabilities of such composite utilization.

This contract is conducted under the sponsorship of the Materials Application Branch of the Materials Division of the NASA Langley Research Center. Mr. H. Benson Dexter, Composite Section, is the NASA project monitor. Mr. W. E. Harvill is the Lockheed-Georgia Program Manager.

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This report is also identified as LG74ER0145 for Lockheed-Georgia Company internal control purposes.

ABSTRACT

One of the most advantageous structural uses of advanced filamentary composites has been shown, in previous studies, to be in areas where selective reinforcement of conventional metallic structure can improve static strength/fatigue endurance at lower weight than that possible if metal reinforcement were used. These advantages are now being demonstrated by design, fabrication, and test of three boron-epoxy reinforced C-130 center wing boxes. This structural component was previously redesigned using an aluminum build-up to meet the increased severity of fatigue loadings. Direct comparisons of relative structural weights, manufacturing costs, and producibility can be obtained, and the long-time flight-service performance of the composite-reinforced structure can be evaluated against the wide background of metal-reinforced structure.

The first three phases of a five-phase NASA program to demonstrate the long-time flight service performance of a selectively reinforced center wing box have been completed. During the first phase of program activity, the advanced development work necessary to support detailed design of a composite reinforced C-130 center wing box was conducted. Activities included the development of a basis for structural design, selection, and verifications of materials and processes, manufacturing and tooling development, and fabrication and test of full-scale portions of the center wing box. Phase I activities have been previously documented in NASA CR-112126.

Phase II activities consisted of preparing detailed design drawings and static strength, fatigue endurance, flutter, and weight analyses required for Phase III wing box fabrication. Some additional component testing was conducted to verify the design for panel buckling, and to evaluate specific local design areas. Development of the "cool tool" restraint concept was completed, and bonding capabilities were evaluated using full-length skin panel and stringer specimens. Phase II activities have been previously reported in NASA CR-112272.

Phase III activities described in this report consisted of the fabrication of three C-130 center wing boxes, selectively reinforced with boron-epoxy composites. Proof tests of the test article have been completed and fatigue testing has been initiated on that wing box under Phase IV program activities. The two flight articles have been fabricated and installed on Air Force C-130 Aircraft Serial No.'s 4557 and 4563 to demonstrate the long-time flight worthiness of advanced composite reinforced aluminum structure.

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ABBREVIATIONS

Abbreviation	Description
AFPRO	Air Force Plant Representative's Office
AFQA	Air Force Quality Assurance
Al	Aluminum
Avg	Average
B/P	Blueprint
$\mathbb{C}_L$	Centerline
CWB	Center Wing Box
DR	Discrepancy Report
FACI	First Article Configuration Inspection
^{o}F	Temperature in degrees fahrenheit
ft	Foot
g	gram
in	Inch
IRAN	Inspect and Repair as Needed
^{o}K	Temperature in degrees Kelvin
kip	One thousand pounds force
ksi	One thousand pound force per square inch
lb	Pound (mass or force)
m	Meter
MRB	Material Review Board
μ	Micro

ABBREVIATIONS (Continued)

Abbreviation	Description
N	Newton (force)
N/m^2	Newton per square meter
P.C. or PC	Process control
PJS	Production job sheets
psi	Pounds force per square inch
SI	International System of Units
W.S. or WS	Wing Station

PROGRAM FOR ESTABLISHING LONG-TIME FLIGHT SERVICE PERFORMANCE OF COMPOSITE MATERIALS IN THE CENTER WING STRUCTURE OF C-130 AIRCRAFT

PHASE III - FABRICATION

By W. E. Harvill and A. O. Kays

1.0 SUMMARY

One of the most advantageous structural uses of advanced filamentary composites is in areas where selective reinforcement of conventional metallic structure can improve static strength/fatigue endurance at lower weight than would be possible if metal reinforcement were used. The first three phases of a five-phase NASA program to demonstrate the long-time flight service performance of a selectively reinforced center wing box have been completed. During the first phase of program activity, the advanced development work necessary to support detailed design of a composite-reinforced C-130 center wing box was conducted. Activities included the development of a basis for structural design, selection and verification of materials and processes, manufacturing and tooling development, and fabrication and test of full-scale portions of the center wing box. Phase I activities have been previously documented in NASA CR-112126, Reference 1.

During Phase II, the basic C-130E aluminum center wing box design was changed by removing aluminum and adding unidirectional boron-epoxy reinforcing laminates bonded to the crown of the hat stiffeners and to the skin under the stiffeners. The laminates were added in a nominal 80/20 area ratio of aluminum to boron-epoxy. Sufficient material was provided to meet ultimate load requirements of the C-130E wing box and the fatigue life of the C-130 B/E wing box.* Laminates are tapered out at the rainbow end fittings and access door openings by progressively stopping individual plies of the tape. Fasteners are used at the ends of the laminates to prevent peeling. Adequate bearing surface was provided in fastener penetration areas by titanium doublers integrally bonded into the laminates. Careful design and manufacturing techniques were used to reduce the number of fasteners (particularly blind fasteners) which penetrate the laminates, thus minimizing potential installation and inspection problems. A total of 129 detailed design drawings were prepared for initiation of the production program.

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*NOTE: The terminology "C-130 B/E" or "B/E" refers to the existing metallic center wing box which is installed in Model C-130B, C-130E, and C-130H aircraft. The C-130H is the designation of the aircraft model currently in production. This aircraft has the metal-reinforced center wing which has been retrofitted to a sizeable part of the total C-130 fleet. The two composite reinforced center wing boxes (flight articles) were installed in C-130H aircraft. In this report, the "B/E" designation always refers to an aircraft model and never means boron-epoxy. Where boron-epoxy is discussed, the words are spelled out.

Detailed substantiating structural, fatigue, and flutter analyses were conducted to assure structural integrity of the reinforced center wing box. Phase II activities are fully reported in NASA CR 112272, Reference 2.

In the third program phase, reported in detail in this document, three composite-reinforced wing boxes were fabricated. After release of the detailed design drawings, manufacturing planning and production paperwork were prepared and released. Materials were purchased and inspected (on receipt) for constructing the three wing boxes. Laminates were laid up, cured, and bonded to the metal adherends in Lockheed's Manufacturing Research Department. Fabrication of metal parts and assembly of the complete wing boxes were accomplished in the normal C-130 production flow at the Lockheed-Georgia Company. Wing surface first-stage assembly in standard production fixtures, illustrated in Figure 1, effectively concludes the portion of assembly peculiar to the use of composite reinforcements. Thereafter, except for minor adjustments due to fastener relocations, all assembly operations were equivalent to those conducted in the normal production of C-130 wing boxes.

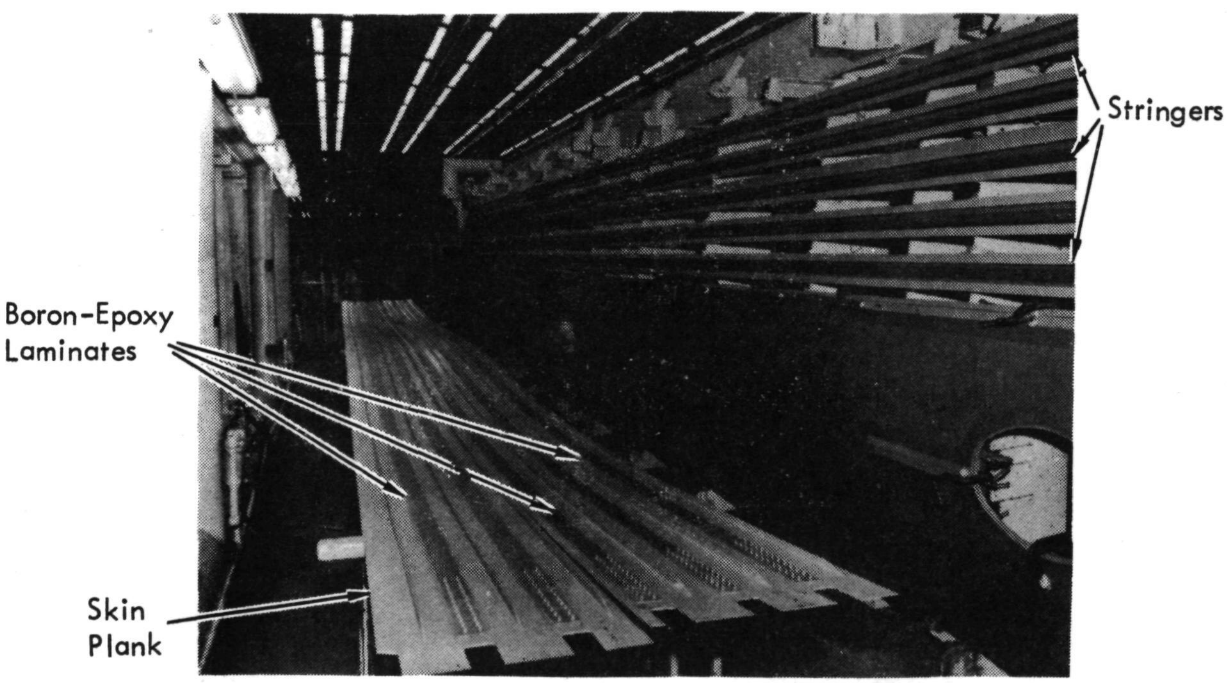


FIGURE 1. - C-130 CWB UPPER SURFACE ASSEMBLY

Throughout the fabrication and assembly activity, thorough inspections were conducted by both Lockheed and Air Force inspectors to assure a high-quality end product. First Article Configuration Inspections (FACI) were conducted on both flight articles to verify that all requirements had been satisfied.

The first flight article was installed in C-130H Serial 4557 (AF73-01592) in June 1974, and the second flight box was installed in Serial 4563 (AF73-01594) in July 1974. These are new aircraft: the center wing boxes were installed in the normal production flow with no difficulties. The first flight article is shown in Figure 2 during installation into Ship 4557.

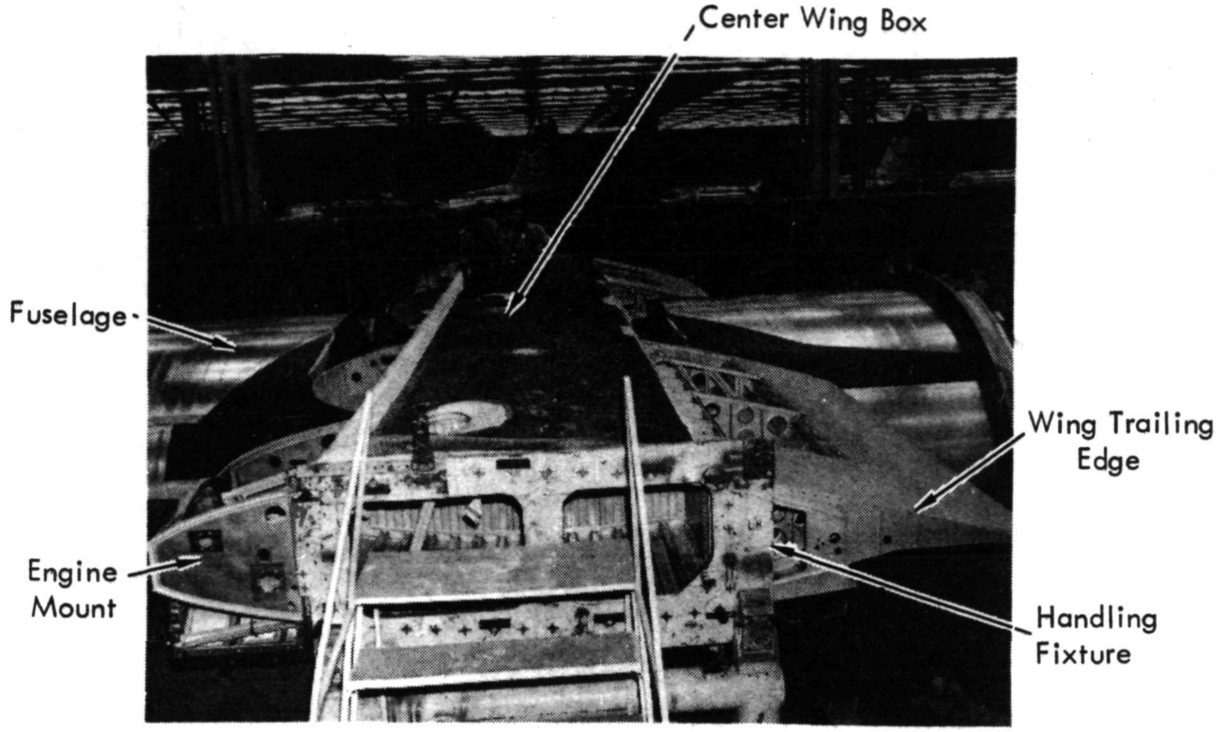


FIGURE 2.- INSTALLATION OF CENTER WING BOX IN C-130 SERIAL 4557

Although weight saving was not a major program goal, and was actually subordinated to accomplishment of flight service program goals, it is, nevertheless, an important factor, and a weight saving of 229 kg (506 lb) was predicted. This prediction, based on calculations from the final production drawings, indicated a saving in total box weight of slightly more than 10 percent. Minor design changes to facilitate production reduced this indicated value to 225 kg (494 lb). Actual weighings of the completed wing boxes showed savings of 222 kg (488 lb) for the test article and 205 kg (450 lb) for the flight articles. These values fall within anticipated manufacturing tolerances. The 318 kg (700 lb) total of boron-epoxy being used in two wing boxes for the 3-year flight evaluation represents a sizeable exposure of boron-epoxy materials to the representative service environment encountered during the life of an aircraft.

It is interesting to note that, in the wing surfaces (where reinforcements were added), an average metal removed/composite added ratio of about 2.5 was achieved. This shows a

high potential for weight saving in future similar applications where less conservative criteria may be established than those used for this particular design.

Cost projections for production quantities of C-130 composite-reinforced center wing boxes were made based on accumulated cost data using an eighty percent cumulative average cost curve. The total cost increase to add boron-epoxy reinforcement is projected for the 200th production wing box to be $40,120 for labor and materials. The computed cost per pound of weight saved is approximately $79.29.

A reliability and quality assurance program was continued in accordance with the approved program plan. The reliability assessment at the end of Phase III is that a high degree of hardware conformance to detail design was achieved. Expectations are high that successful test and service experiences can be realized with a minimum of difficulty.

The successful completion of the composite-reinforced center wing boxes enabled initiation of the test phase of the program. Ground acceptance tests have begun and are proceeding satisfactorily. Both upbending and downbending proof (limit) load static tests have been completed and fatigue testing to four lifetimes is underway. The test article, photographed while sustaining limit downbending design load, is shown in Figure 3.

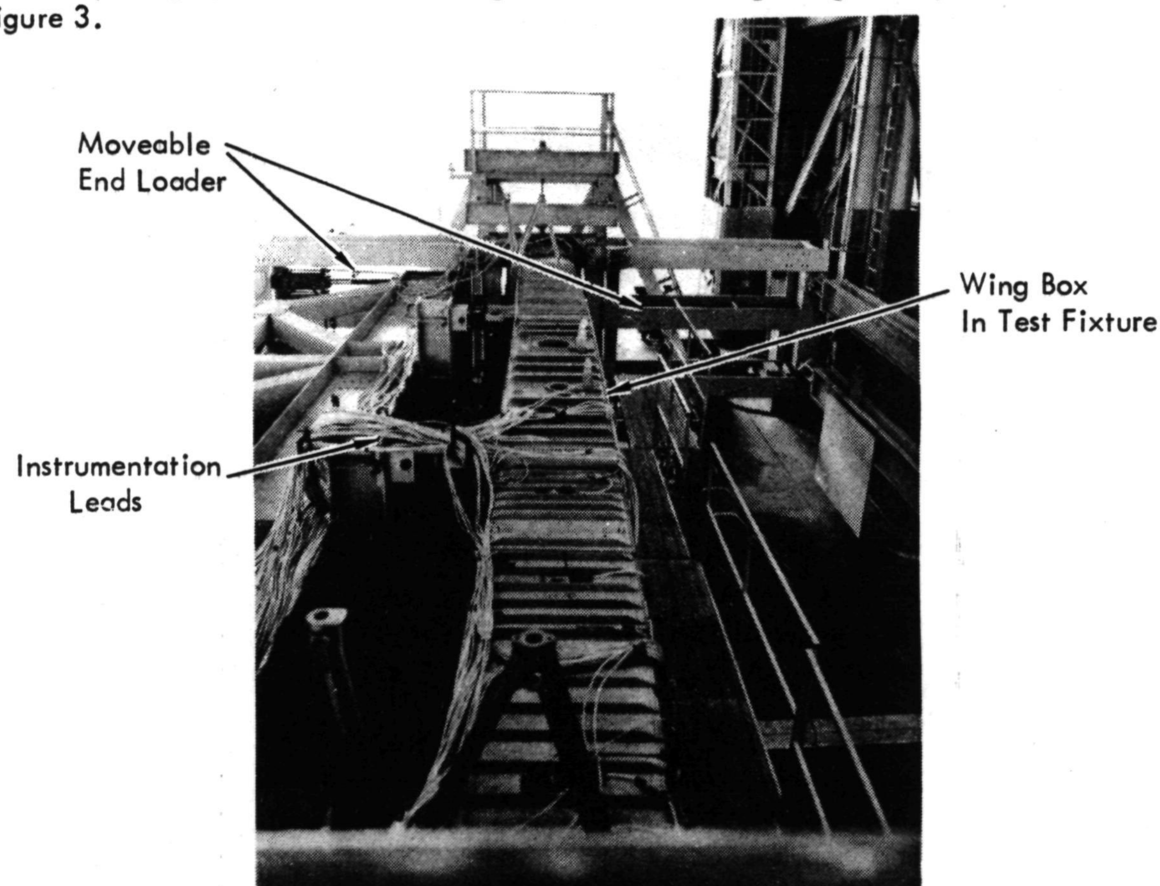


FIGURE 3. - LIMIT LOAD DOWNBENDING TEST

2.0 INTRODUCTION

Application studies and Advanced Development tests (Reference 3 and 1, respectively), conducted for NASA by Lockheed, have shown that boron-epoxy composite laminates bonded to the skin and stiffeners of the C-130 aircraft center wing box can significantly improve the overall fatigue endurance of the structure, at a lower weight than that possible if metal reinforcements were used to achieve the same endurance levels. These advantages will be demonstrated by designing, fabricating, and testing three boron-epoxy reinforced C-130E center wing boxes, in a five-phase program extending over 5-1/2 years. The program phases and associated schedules are illustrated in Figure 4. Phase I, II, and III have been completed. Documentation of activities is included in this report and in References 1 through 3.

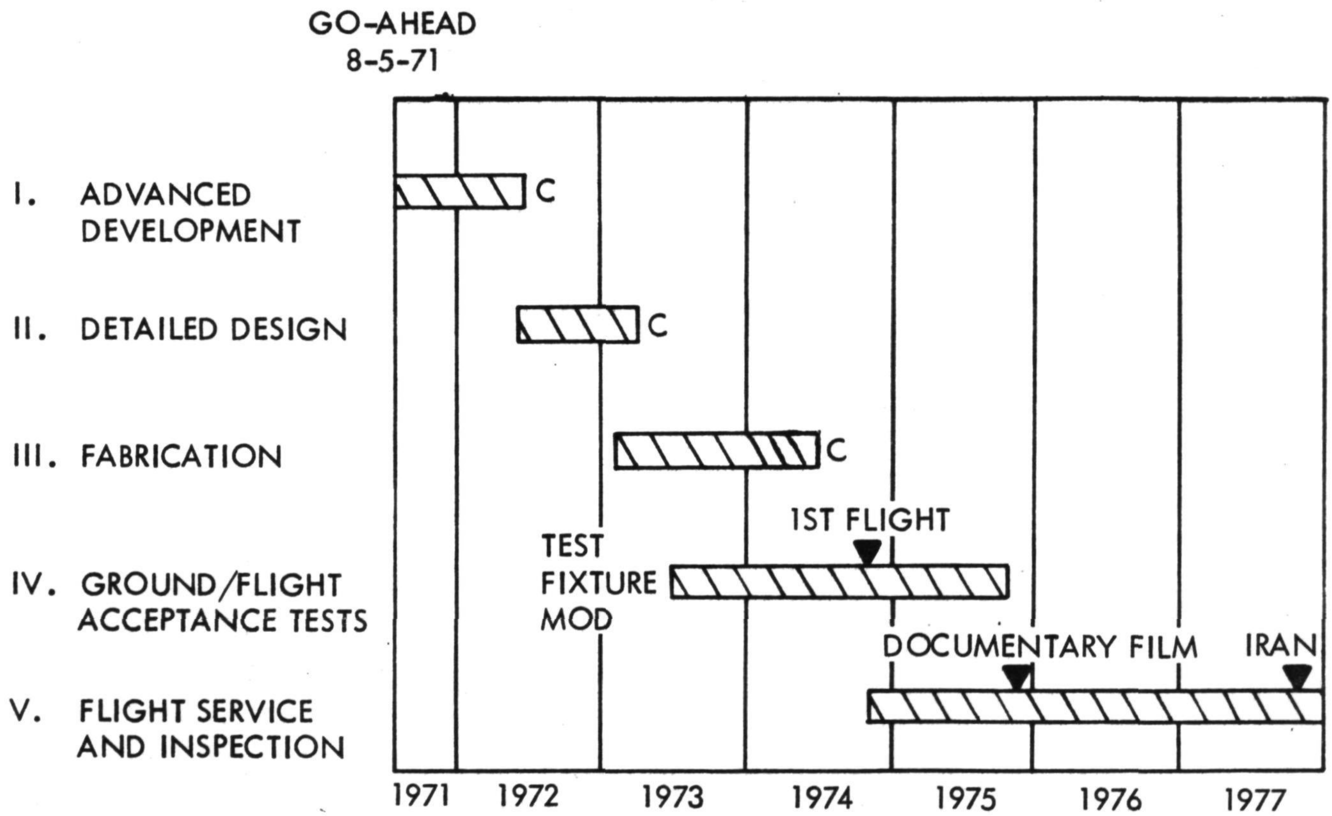


FIGURE 4. - SCHEDULE

The center wing box size and location are illustrated in Figure 5. The box is 11.2m (440 in.) long, 2.03m (80 in.) in chord and, in the all-metal configurations, weighs about 2243 kg (4944 lb). The all-metal configuration is illustrated in Figure 6.

During Phase I, the advanced development work necessary to support detailed design of a composite reinforced C-130 center wing box was conducted. Activities included the development of a basis for structural design, selection of materials and processes, manufacturing and tooling development, and fabrication and test of full-scale portions of the center wing box. The Phase I results further confirmed that, with boron-epoxy reinforcements as shown in Figure 7, equivalent static strength and fatigue endurance could be provided with a significant weight savings. The aluminum skins and stringers have thicknesses less than those of the existing metallic center wing box in Model C-130 B/E aircraft. Equivalent strength is provided by the unidirectional composite.

Phase II activities consisted of preparing detailed design drawings and conducting the substantiating static strength, fatigue endurance, flutter, and weight analyses required for proceeding into Phase III wing box fabrication. Some additional component testing was conducted to complete the panel buckling evaluation and to evaluate specific local design concepts. Tooling development activities were continued to further refine the "cool tool" concept and to evaluate residual stresses with full-length skin panels and stringers. The final design configuration is structurally and functionally interchangeable with the production C-130 B/E wing box.

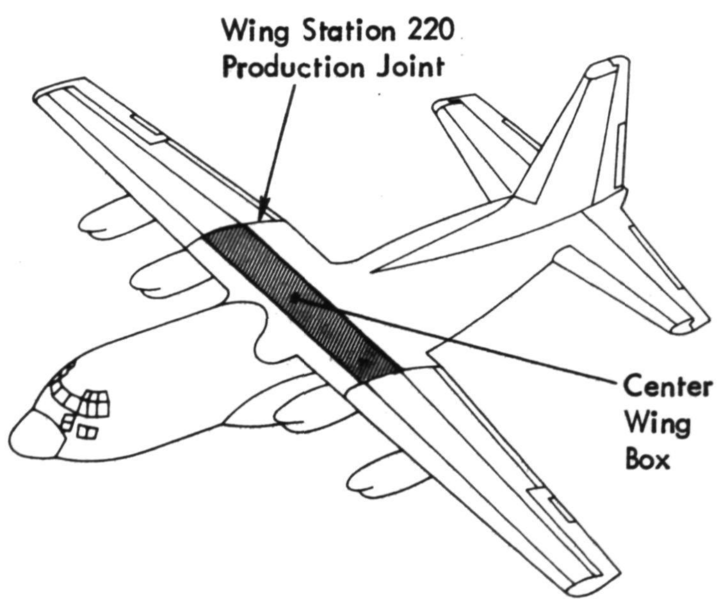


FIGURE 5.- C-130 CENTER WING BOX LOCATION

1. UPPER SURFACE PANELS
2. UPPER SURFACE STRINGERS
3. UPPER SURFACE RAINBOW FITTING
4. LOWER SURFACE PANELS
5. LOWER SURFACE STRINGERS
6. LOWER SURFACE RAINBOW FITTING
7. FRONT BEAM
8. REAR BEAM

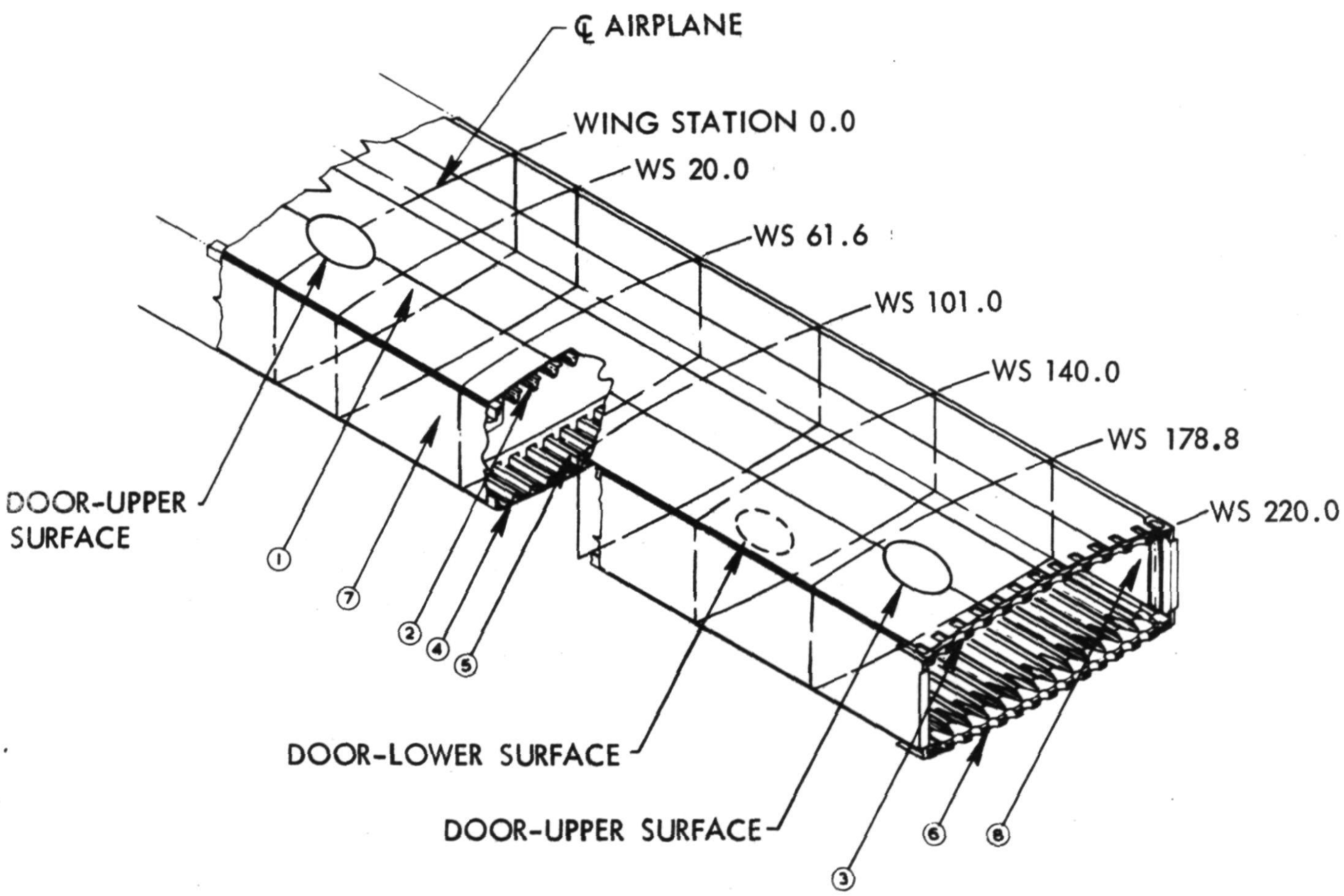


FIGURE 6.- MODEL C-130 B/E CENTER WING BOX

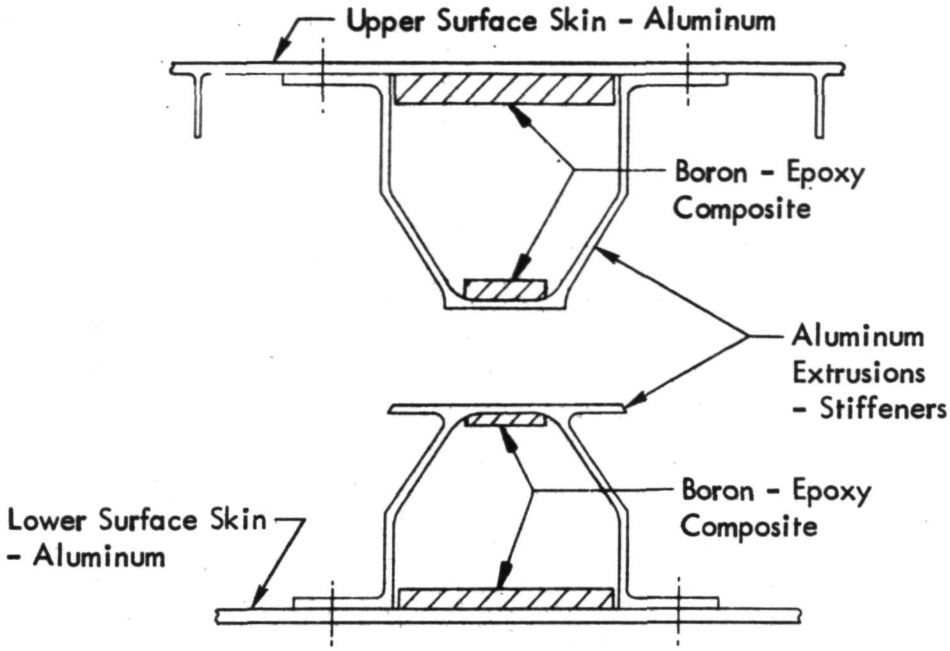


FIGURE 7.- COMPOSITE REINFORCEMENT CONCEPT

Phase II activities consisted of preparing detailed design drawings and conducting the substantiating static strength, fatigue endurance, flutter, and weight analyses required for proceeding into Phase III wing box fabrication. Some additional component testing was conducted to complete the panel buckling evaluation and to evaluate specific local design concepts. Tooling development activities were continued to further refine the "cool tool" concept and to evaluate residual stresses with full-length skin panels and stringers. The final design configuration is structurally and functionally interchangeable with the production C-130 B/E wing box.

In Phase III, reported herein, fabrication and assembly of three composite-reinforced wing boxes was completed. After a joint USAF-NASA-Lockheed configuration review, two of these boxes were released for installation in two Air Force C-130H aircraft to be used in regular operational service by the Tactical Air Command. Service experience will be monitored and documented. Detailed inspections of these two wing boxes, including the use of sophisticated non-destructive test techniques, are scheduled to coincide with regularly phased aircraft inspections.

The first composite-reinforced wing box has been static tested to limit load, and is now being endurance tested to a fatigue spectrum representative of four aircraft lifetimes. Finally, this box will be tested statically to determine its residual strength.

3.0 MATERIALS

New material development for this program was minimal, and was limited to adhesives and their processing. This development work, conducted in Phase I, provided a low-temperature, 386 ±8.3°K (235 ±15°F), curing adhesive system for bonding boron-epoxy laminates to aluminum. All other materials, such as boron-epoxy preimpregnated tape, aluminum, sealants, finishes, titanium, and fasteners were procured and/or processed to the requirements of existing Lockheed specifications.

The boron-epoxy prepreg selected for use in making reinforcing laminates was AVCO Rigidite 5505/4. Tape was purchased, to the requirements of material specification STM 22-450, in net widths to eliminate slitting, and was provided by the supplier in a very timely manner. Table I summarizes the boron-epoxy tape production requirement and delivery schedule for both the 2.29 cm (0.9 in.) wide and the 5.08 cm (2.0 in.) wide tapes. Lot acceptance data are shown in Table II, where each value is an average of three test data points.

The aluminum "planks" for wing skins were machined from existing C-130 B/E extrusions drawn from stock. The hat-section stringers were purchased especially for this usage and were of C-130 B/E material in the lighter C-130E sizes, to reduce the stringer machining required. The skins are of 7075-T73 material and the stringers are mixed: some 7075-T73 and some 7075-T6. These parts are the same material as that in use on production C-130 center wings. Only the wing "covers" were affected by the laminate reinforcement: all other parts (ribs, spars, beam caps, etc.) are standard C-130 production parts.

Two materials-related difficulties were experienced. In the first shipment of 5.08 cm (2.0 in.) tape, the tape was not centered on its release film, resulting in improper tape positioning during lay-up. The second involved low lap shear data from process control specimens which were made in early laminate autoclave runs to verify the bond of interleaved titanium shims. It did not involve the primary bonding adhesive, AF-127-3. This was shown to be peculiar to the test specimen and was related to an inherent high flow in the adhesive. An equally qualified epoxy adhesive, EA9601, was substituted for the remainder of the program, since it provided finger panel shear data more representative of values achieved in the actual bond. Both of these problems were thoroughly investigated. They are discussed in detail in Section 6.0 of this report.

TABLE I.- BORON-EPOXY TAPE REQUIREMENT AND DELIVERY SCHEDULE

5.08 cm (2.0 in.) Wide Tape					
Quantity, m (ft)		Receipt Date		Identification	
Ordered	Received	Scheduled	Actual	Supplier	Lockheed
2438 (8,000)	2438 (8,000)	4/17/73	4/3/73	Lot 59	86430
9449 (31,000)	9461 (31,040)	5/15/73	5/2/73	Lot 59	87331
9449 (31,000)	9481 (31,105)	6/15/73	6/6/73	Lot 59, 60	87904, 87905
9449 (31,000)	9507 (31,190)	6/29/73	6/26/73	Lot 59, 60	88849

2.25 cm (0.885 in.) Wide Tape					
Quantity, m (ft)		Receipt Date		Identification	
Ordered	Received	Scheduled	Actual	Supplier	Lockheed
3,658 (12,000)	3,656 (11,995)	4/17/73	4/3/73	Lot 59	86431
10,973 (36,000)	10,973 (36,000)	5/15/73	5/2/73	Lot 59	87332
10,973 (36,000)	11,042 (36,230)	6/15/73	6/6/73	Lot 59, 60	87906, 87907
10,973 (36,000)	10,973 (36,000)	6/29/73	7/6/73	Lot 60, 61	88847, 88848

TABLE II. - BORON-EPOXY LOT ACCEPTANCE TEST RESULTS

	Shipment No.	Lot No.	Lockheed Control No.	0° Flexural Strength		Shear Strength	
				(GN/m^2)	KSI	MN/m^2	KSI
5.08 cm (2.0 in.) Wide Tape	1	59	86430	1.88	273	101	14.7
	2	59	87331	2.01	291	97.2	14.1
	3	59	87904	1.95	283	106	15.4
		60	87905	1.89	274	106	15.4
	4	60	88849	1.84	267	100	14.5
2.25 cm (0.885 in.) Wide Tape	1	59	86431	1.66	241	88.3	15.0
	2	59	87332	1.78	259	99.3	14.4
	3	59	87906	2.03	294	103	14.9
		60	87907	1.81	263	93	13.5
	4	60	88847	1.99	288	105	15.3
		61	88848	1.62	235*	86.9	12.6*
Specification Requirement				1.65	240	89.6	13.0

*Initial tests showed flexural strength for this lot to be 2% below specification requirement and shear strength to be 3% below specification. This lot was later accepted based on recertification values of 2.07 GN/m^2 (300 psi) for the 0° flexural strength and 101 MN/m^2 (14.7 psi) for the shear strength.

4.0 MANUFACTURING

Three C-130 center wing boxes, selectively reinforced with boron-epoxy composites, were fabricated during the manufacturing phase. The first of these, the test article, has been successfully static proof tested and fatigue tests simulating four aircraft lifetimes have been initiated. The two flight articles have been installed on Air Force C-130H aircraft to begin a three-year flight service evaluation.

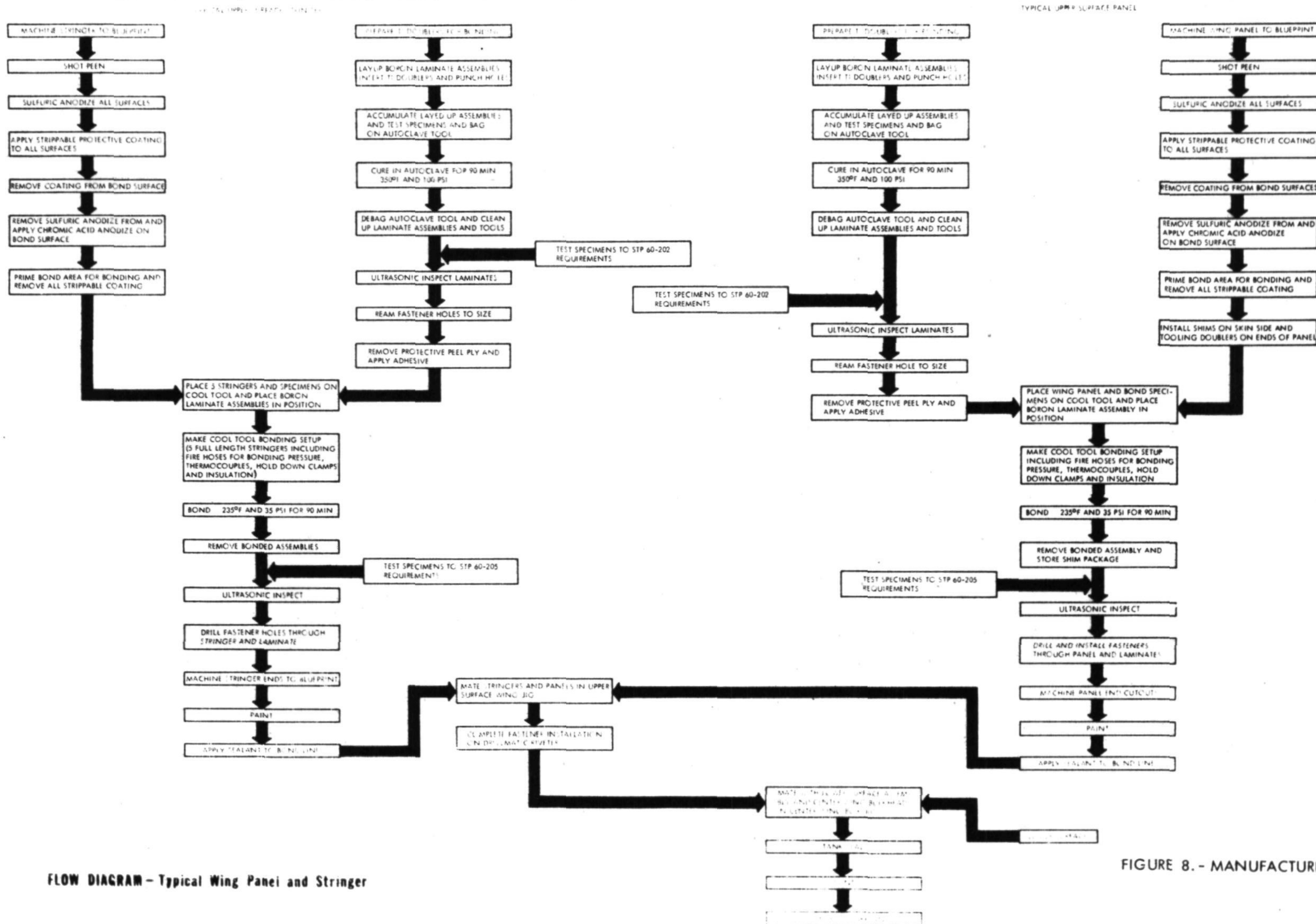
The three center-wing boxes were built in conformance with the basic design established during Phase II. This basic design consisted of the following:

> Reinforcement of the upper and lower surface assemblies of the wing box was accomplished by designing new skin panels and hat-section stringers and adding boron-epoxy laminates. The cross-sectional area of the aluminum C-130B/E skin panels and hat-section stringers was reduced to the original C-130E cross-sectional area. Access door areas, wing station 220 joint rainbow fittings, and splice straps were retained in the heavier C-130B/E configuration. Skin panels and stringers were taper-transitioned to the thinner C-130E configuration inboard of the joint fittings and on each side of the access doors. Outboard of the upper surface outboard access doors, the C-130B/E configuration was retained because of the close proximity of the W.S. 220 joint. Other sub-components of the existing model C-130B/E wing box, such as ribs, spars, fittings, brackets, and access doors, were not changed unless a relocation was required to eliminate holes in the reinforcing laminate.

4.1 MANUFACTURING PLAN

A comprehensive manufacturing plan was developed for production of the wing boxes. Detailed shop orders were written for fabrication of all metal parts, composite parts, bonded assemblies, and other subassemblies. Production Job Sheets (PJS's) were written for the wing box assembly. The PJS's included both a listing of parts required and a detailed sequence of assembly operations. As each shop order or PJS is completed it is reviewed for compliance to all applicable requirements and is "stamped off" by both the completing department and by the cognizant quality assurance inspector. These papers are then filed in inspection records as permanent documentation.

The manufacturing plan is illustrated in flow chart form in Figure 8. The step-by-step sequence for these operations is described in Appendix B. The manufacturing plan shows the flow for a typical upper surface stringer and an upper surface panel. These sequences are identical to those used for the other stringers and panels. The manufacturing process is described in the following sub-sections of this report:



FLOW DIAGRAM – Typical Wing Panel and Stringer

FIGURE 8. - MANUFACTURING PLAN

- o Section 4.2 describes tooling for laminate fabrication and bonding.
- o Sections 4.3 - 4.5 describe adherend fabrication and bonding.
- o Section 4.6 covers the center wing box assembly.

4.2 TOOLING

The layup and cure of the boron-epoxy laminates, and the subsequent bonding of the laminate-to-metal parts, required a number of tools as aids to perform these operations. This tooling consisted of (1) drill templates and punch tools, (2) laminate layup and cure tools, and (3) laminate-to-metal bonding fixture.

4.2.1 Drill Templates and Punch Tools

Drill templates to generate proper hole spacing in the titanium shims were fabricated for the shims in each of the laminates. These tools consisted of 1.27 cm (0.5 inch) thick aluminum blocks with locating pins to align the titanium shims and the close tolerance drill bushings. The tools were coordinated to both the punch tools and the laminate layup and cure dams to minimize the possibility of errors in hole patterns. A total of twenty drill tools were used. An exploded view of a typical drill tool is shown in Figure 9.

An exploded view of the companion punch tool for punching the holes undersize in the uncured laminate is shown in Figure 10. The punch tools were aligned relative to the dams by locating pins which, in turn, align the laminate holes with the titanium shim holes, and with clearance holes in the bottom of the dams. The layup technician, using a steel punch and hammer, punches the hole undersize in the uncured boron-epoxy laminate, as shown in Figure 11, through the bushing so that the titanium is not touched.

Due to the difference in the allowable out of the freezer times between the uncured boron-epoxy and the adhesive, which is placed on the titanium doublers, it was sometimes necessary to punch the boron-epoxy before inserting the titanium shims in the laminate. In these cases, a piece of tape release paper is placed in the laminate as a "stand-in" for the titanium and the punching operation was performed as if the titanium were in place. Later, the release paper was replaced with the adhesive-coated titanium shims. As illustrated in Figure 12, pins are used to coordinate the hole alignment.

4.2.2 Laminate Layup and Cure Tools

A tape-dispensing device, illustrated in Figure 13, was designed and fabricated as an aid in laying the boron-epoxy laminate plies in their respective dams. The tape-laying device consisted of a tape-dispensing reel, a separator film take-up reel, a tape guide and alignment roll, a spring-loaded tape compressing roll, and an angled guide track. All

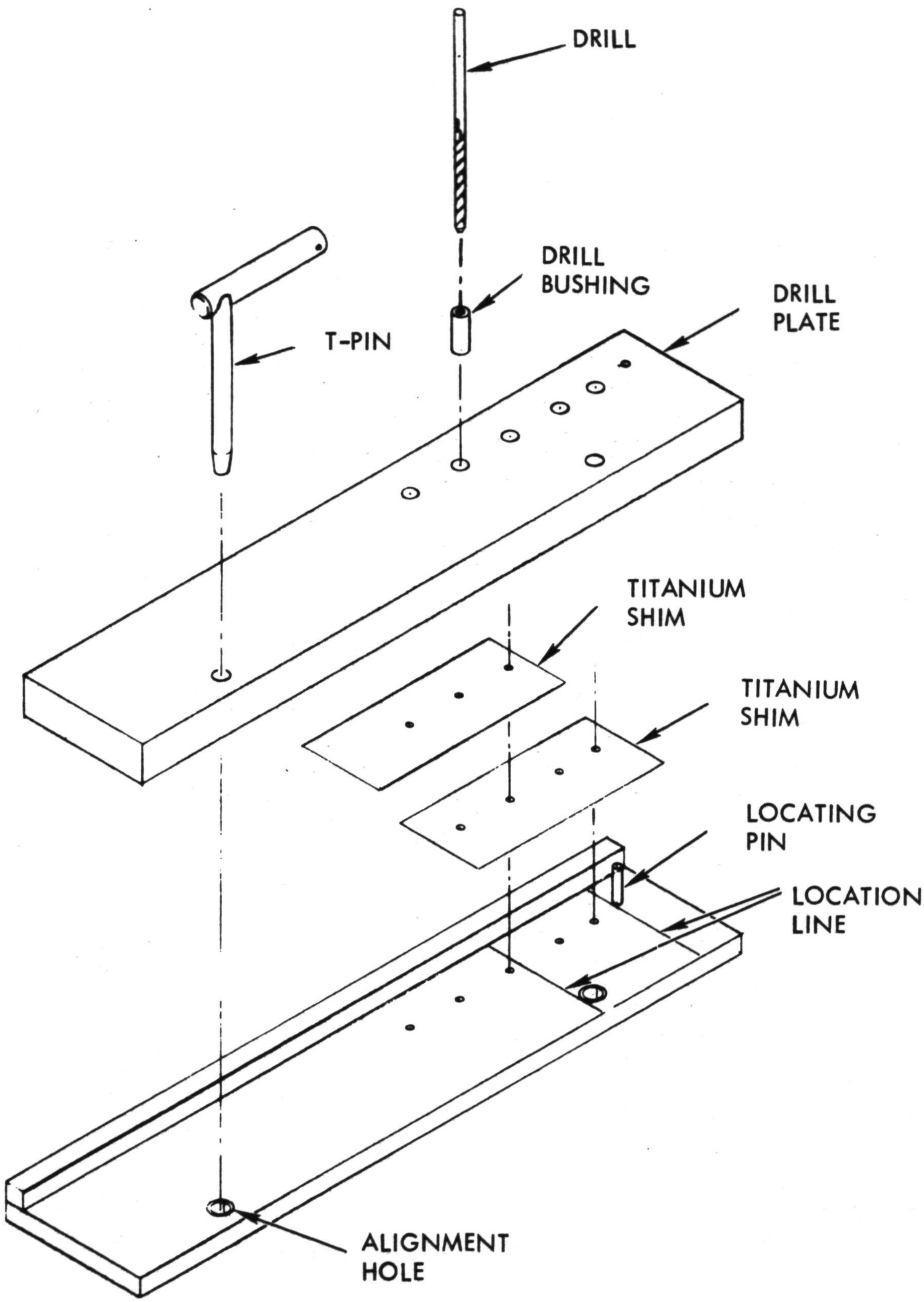


FIGURE 9. - TYPICAL DRILL TOOL FOR DRILLING HOLES IN TITANIUM SHIMS

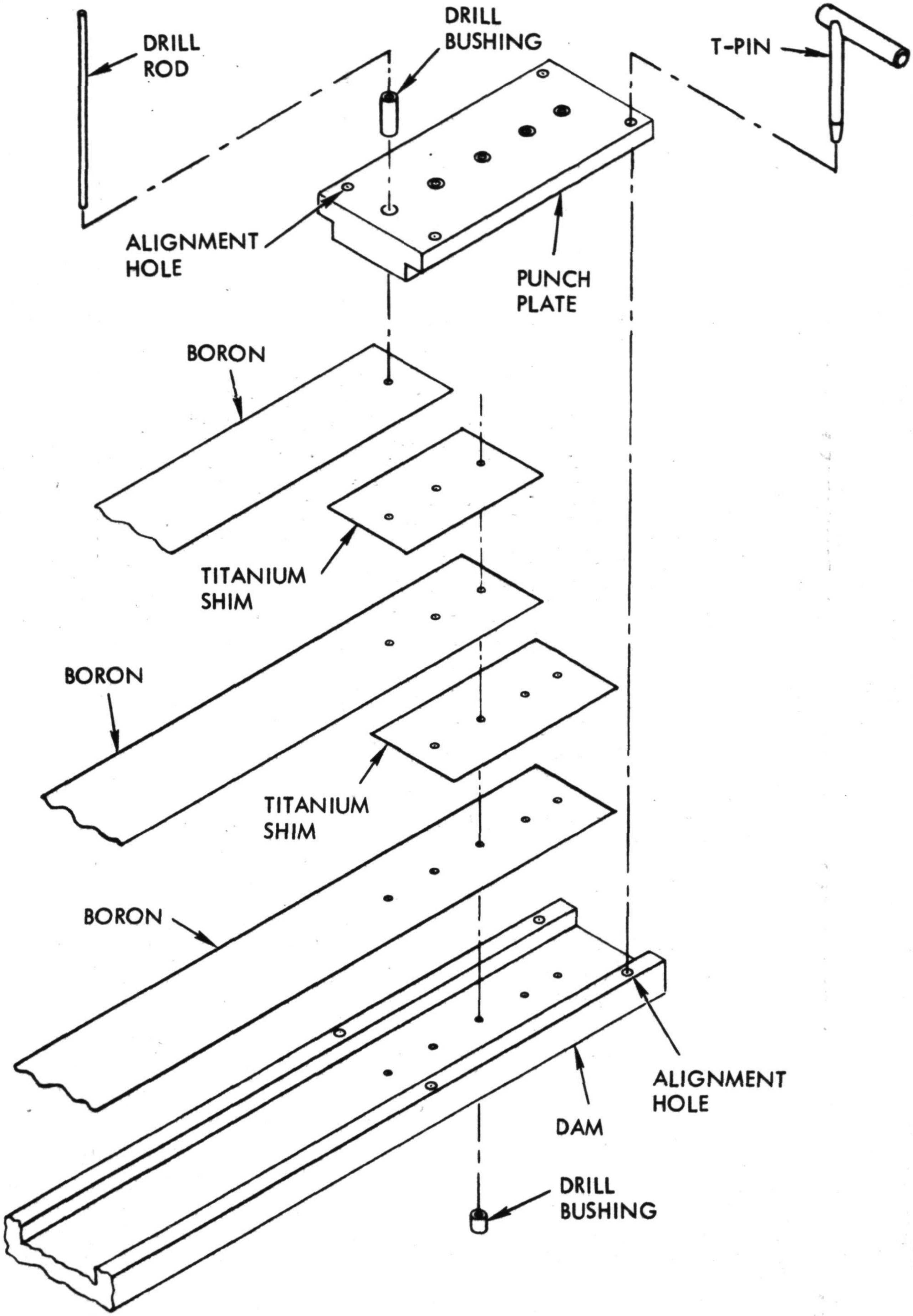


FIGURE 10.- TYPICAL PUNCH TOOLING FOR PUNCHING HOLES IN UNCURED BORON-EPOXY LAMINATE

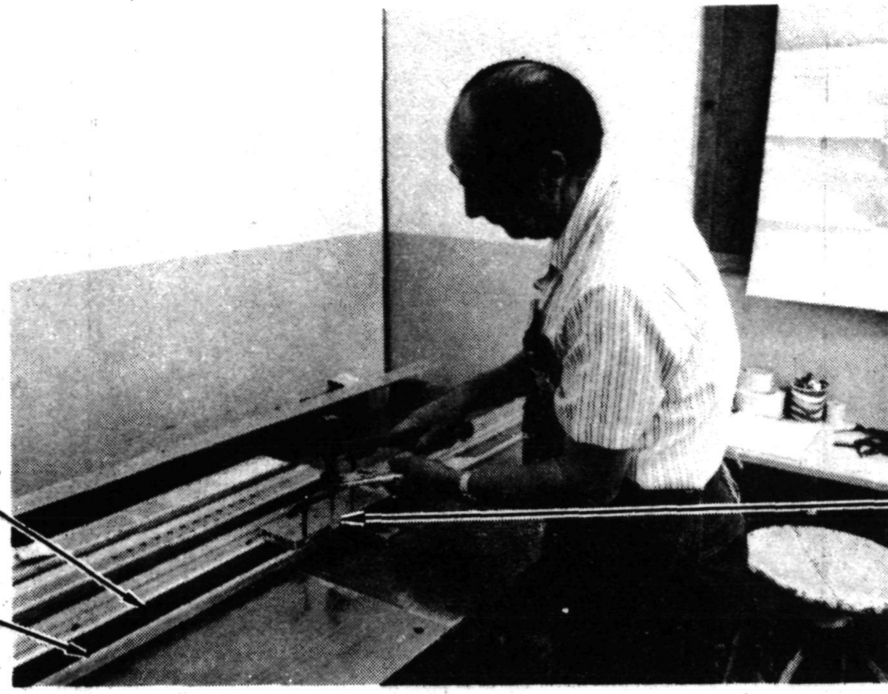


FIGURE 11. - PUNCHING HOLES THROUGH BORON-EPOXY AND PREDRILLED TITANIUM SHIMS

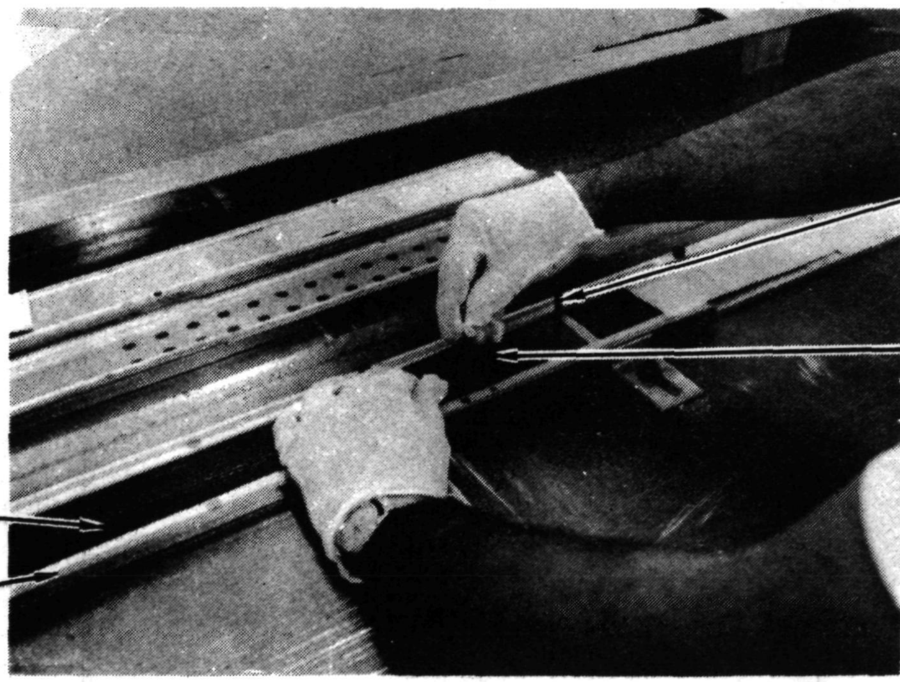


FIGURE 12. - LOCATING PREDRILLED TITANIUM SHIM

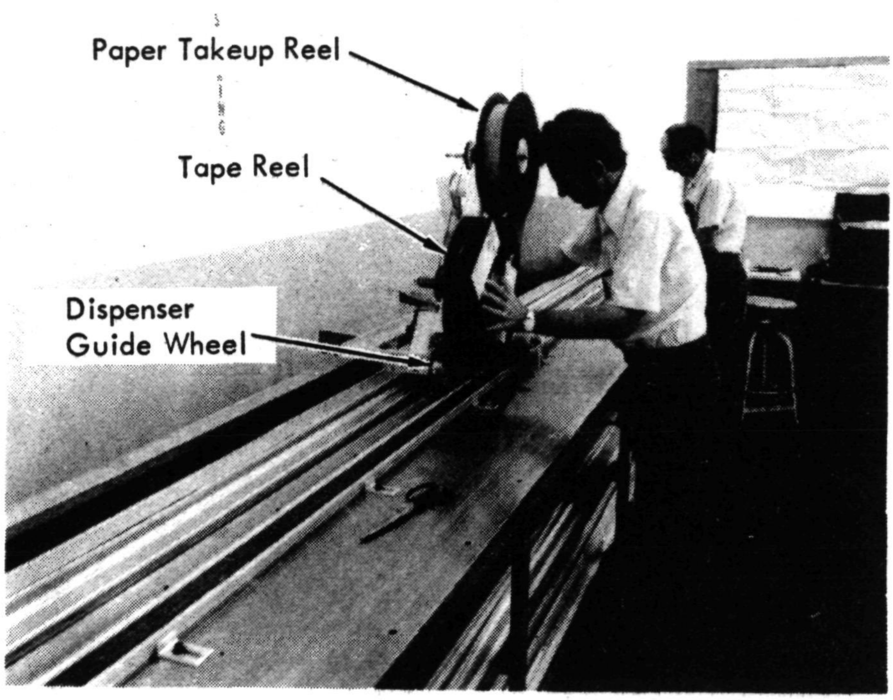


FIGURE 13. - TAPE DISPENSING DEVICE

were mounted on a moveable trolley on a 40-foot long aluminum table, shown in Figure 14, where the tape dispenser is seen in the extreme background. Figures 15 and 16 show the tape-dispensing device being used to place the boron-epoxy tape in its dam.

Individual metal dams were made for each laminate required. The dams were constructed from 1/4 inch thick aluminum base plates with 1/4" x 1/2" cross-section side rails (dams). Stacked multiple dams are used when necessary to accommodate increased laminate thickness. Figure 17 shows the drilling operation for pinning and fastening the dams to the base to form the mold for a stringer laminate. One side of the dam has been installed and the second side is being drilled. The gap between the two sides of the dam was maintained by a gage block. The drill templates, discussed earlier, were used for locating the pin patterns in the boron-epoxy laminate molds, as shown in Figure 18. These pins maintain the pre-punched holes in the boron-epoxy tape during cure and also maintain the proper position for the internal titanium doublers. The completed holes with pin bushings installed are shown in Figure 19. In addition, ply drop-off position marking and identification can be seen in Figure 19.

The autoclave tool for curing laminates was a standard 40-foot tool on wheels, modified to provide booster electrical heat to enable attainment of higher temperatures than those available from the present autoclave heat system. Controller-operated, zoned, fiberglass/nichrome heaters were placed on insulation boards on the tool face. An aluminum plate was placed on the top of the heater blankets to spread the heat evenly into the laminates. The advantage of placing the heater blankets against the aluminum plate instead of directly on the autoclave tool was that the heat-up was more consistent and immediate at the bondline instead of first having to heat up the entire autoclave tool. This temperature was continuously monitored with thermocouples. The heater blankets were made up in six zones to allow accurate control of the heat and heat-up rate from outside the autoclave.

Attached to the aluminum plate is a guide bar (Figure 20) to align the laminate dams. A second bar runs along the opposite side with screws in it to act as a clamp holding the dams against the guide bar. Figure 20 illustrates the finished construction method of this tool except for the bagging materials. The boron-epoxy laminates, in their individual molds were transferred directly from the tape lay-up table to this tool and transported to the autoclave for curing.

4.2.3 Laminate-to-Metal Bond Fixture

Achievement of a low-stress bondline between the laminate and metal adherends is important, both in attaining extended fatigue endurance from the reinforced components, and in minimizing subsequent assembly sequences. Differential stresses are created by the differential thermal expansions of the two adherends during the elevated temperature bonding cycle. The aluminum adherend has a coefficient of thermal expansion approximately five times that of the unidirectional boron-epoxy laminate. This difference can cause high residual stresses at temperatures other than the bonding cure temperature. Development of bonding techniques to minimize this problem was a sizeable part of the Phase I activity, and culminated in the "cool tool" restraint process, previously reported in NASA CR 112126.

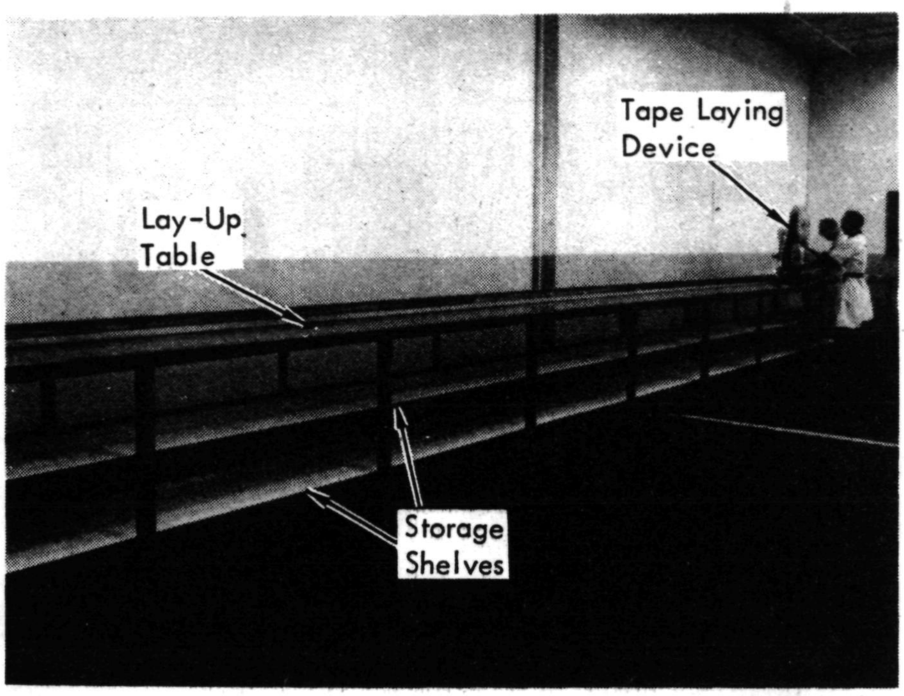


FIGURE 14. - FORTY FOOT LONG BORON-EPOXY LAYUP TABLE: TAPE LAYING HEAD SHOWN BEING ADJUSTED ON RAILS

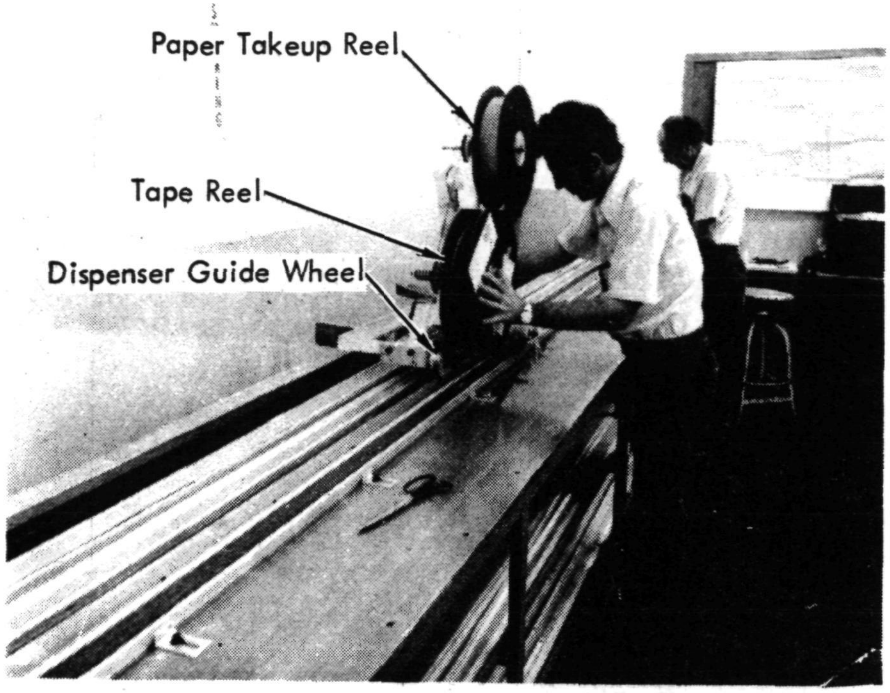


FIGURE 15. -TAPE DISPENSING DEVICE

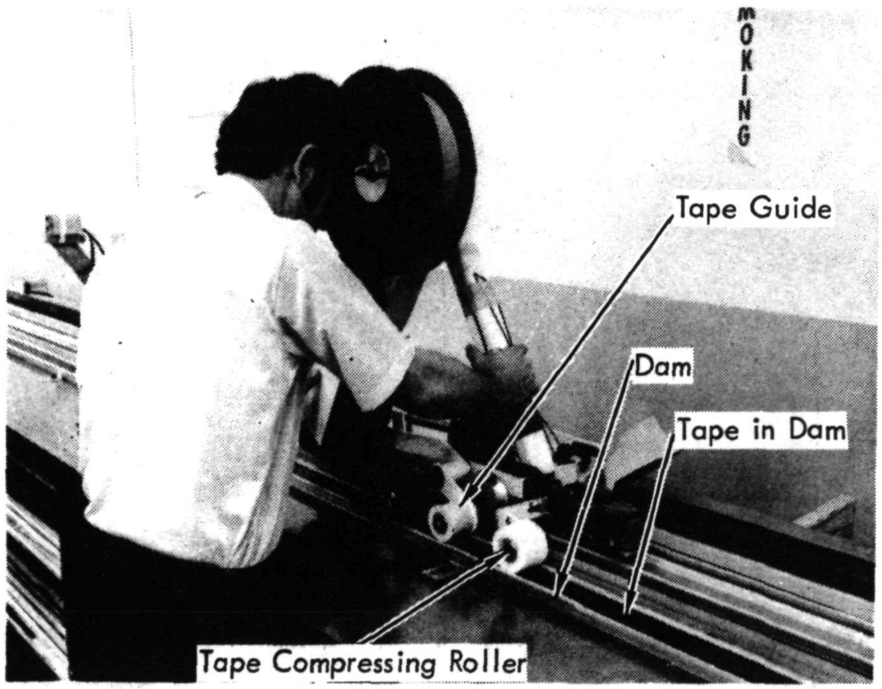


FIGURE 16. - TAPE DISPENSING HEAD LAYING BORON-EPOXY TAPE IN DAM

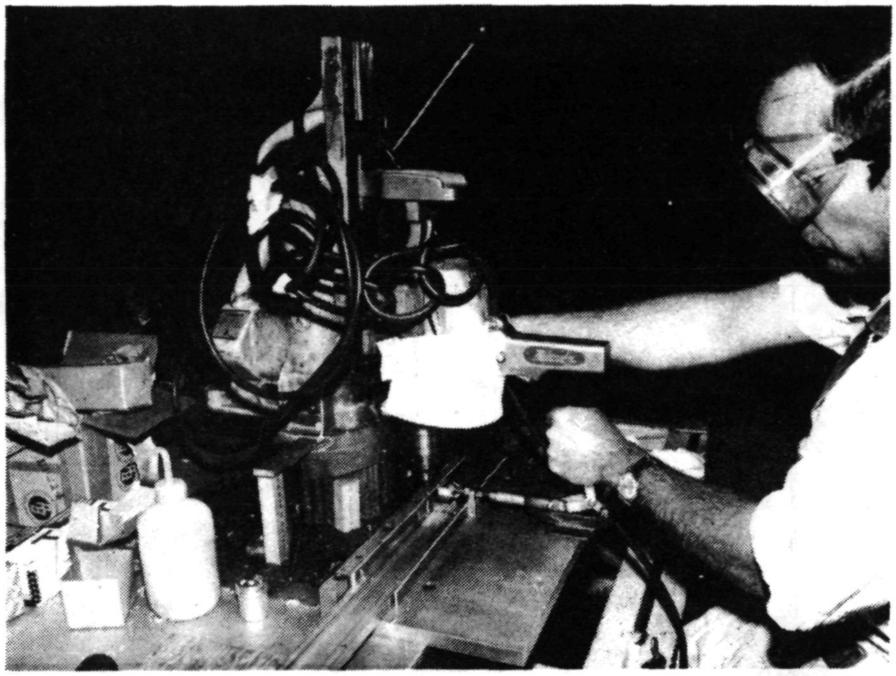


FIGURE 17.- ASSEMBLY OF DAMS FOR BORON-EPOXY LAMINATES

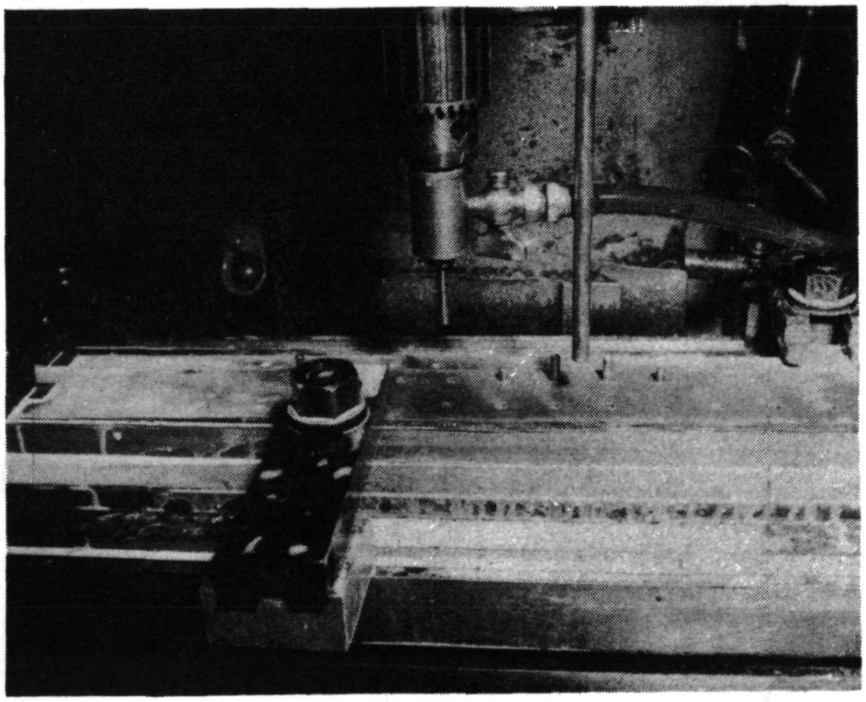


FIGURE 18.- TRANSFERRING HOLES FROM DRILL TEMPLATE TO DAMS

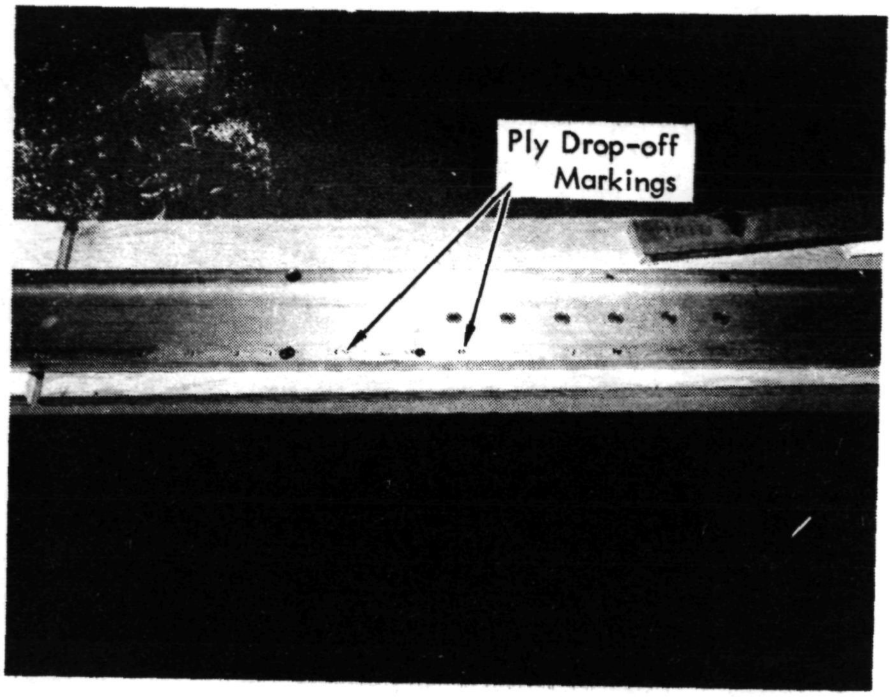


FIGURE 19.- TYPICAL VIEW OF LAMINATE CURE DAMS IN AREA OF MECHANICAL FASTENERS

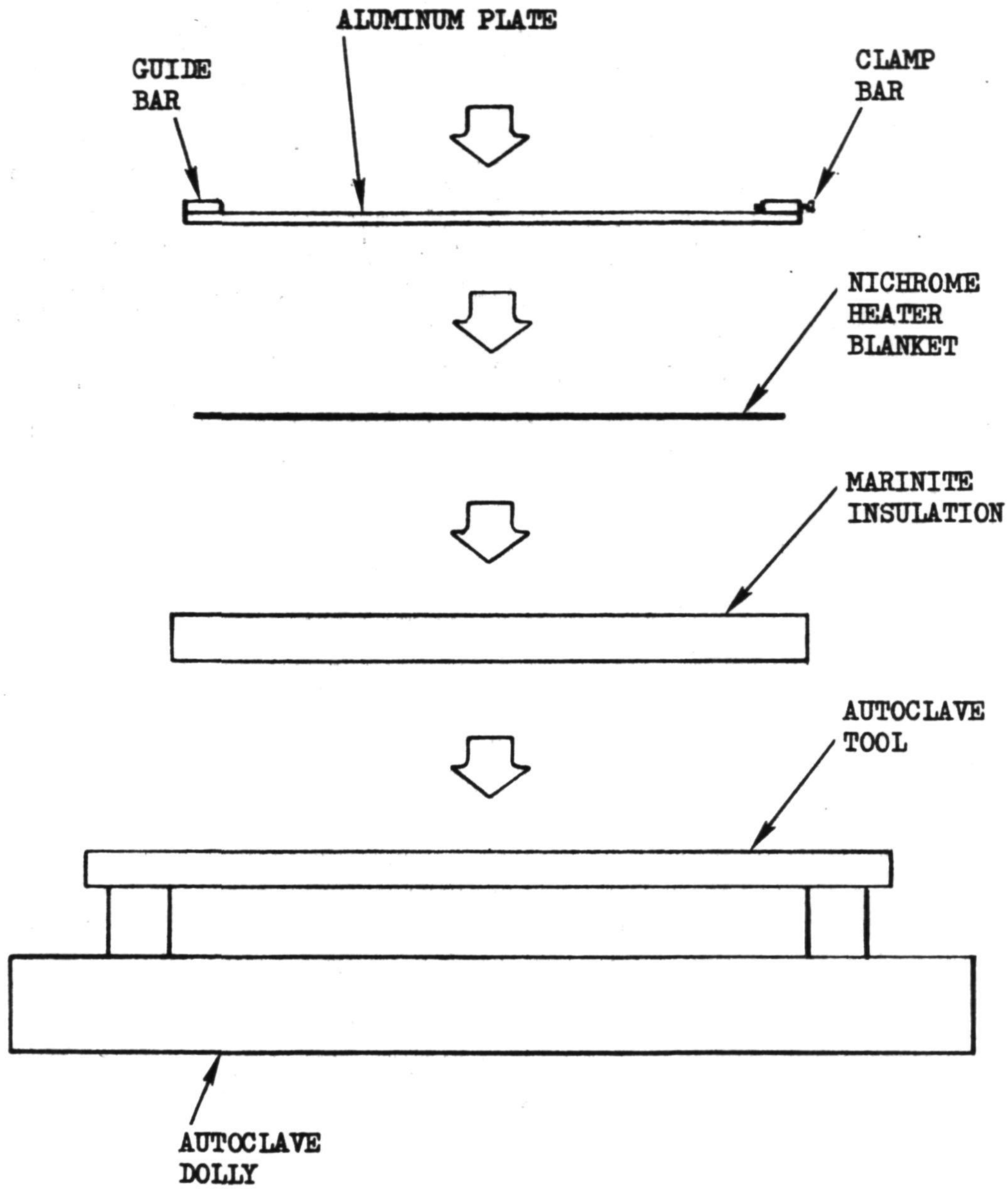


FIGURE 20. - LAMINATE CURING TOOL (CROSS-SECTION SKETCH)

A cross section sketch of the "cool tool," used for restraining the metal parts during the bonding cycle, is shown in Figure 21. It consists of steel beams welded to a steel face sheet overlaid with insulation. Two layers of 181 fiberglass cloth were placed over the insulation to provide additional insulation for the fiberglass/nichrome heater blanket and for the steel tool. The zoned heater blanket is covered with a teflon film slip sheet, followed by a thin rubber electrical insulator. A second film slip sheet is added and the tool buildup is completed by a partitioned aluminum sheet. The aluminum sheet is not continuous over the tool length, but is sectioned, with gaps to allow for expansion when heat is applied. Figure 22 shows the basic steel tool.

End restraints for the tool were provided to prevent expansion of the metal adherend. These restraints were sized to withstand the maximum thermal stresses produced, with a minimum factor of safety of 2.5. Details of the restraint are illustrated in Figure 23.

4.3 METAL PARTS FABRICATION

The metal parts which were affected by the boron-epoxy reinforcement were the wing skin panels and hat section stringers. All other parts, such as beams, joint fittings, ribs, and doors, were standard C-130 production parts manufactured under existing C-130 controls. They were called up by stock number from stores when needed for assembly.

The skin panels and stringers were machined from basic extrusions to the required configuration. Machining was accomplished on numerically controlled milling machines and the parts were subsequently shot-peened. The panels and stringers were then sulfuric acid anodized, spray-coated with chem-mill maskant, and stored until ready for bonding. When needed for the bonding operation, they were removed from stock and the bond areas were prepared for reduced chromic acid anodizing by removing the chem-mill maskant from the bond areas. After anodizing, the bondline area was primed for protection and the remaining maskant was stripped prior to loading the parts into the bond fixture.

4.3.1 Machining

The upper and lower surface panels and stringers were machined by numerically controlled machines to the required thicknesses and to closely controlled aerodynamic tapers in accordance with the applicable drawing. Figures 24 and 25 indicate the amount of material which had to be machined from the panels. Figure 26 shows a stringer being machined.

Although stringer machining was time-consuming, it was essentially trouble-free. Panel machining presented a more difficult problem, since these planks are sculptured from end-to-end and (on some panels) in the chord direction as well. New NC tapes were required (one for each panel -- seven in all) and a full panel tool try of the applicable tape was run for each plank. Subsequently, nine panels were scrapped due to improper tooling and/or workmanship errors, resulting in a ratio of 70% for finished panels accepted

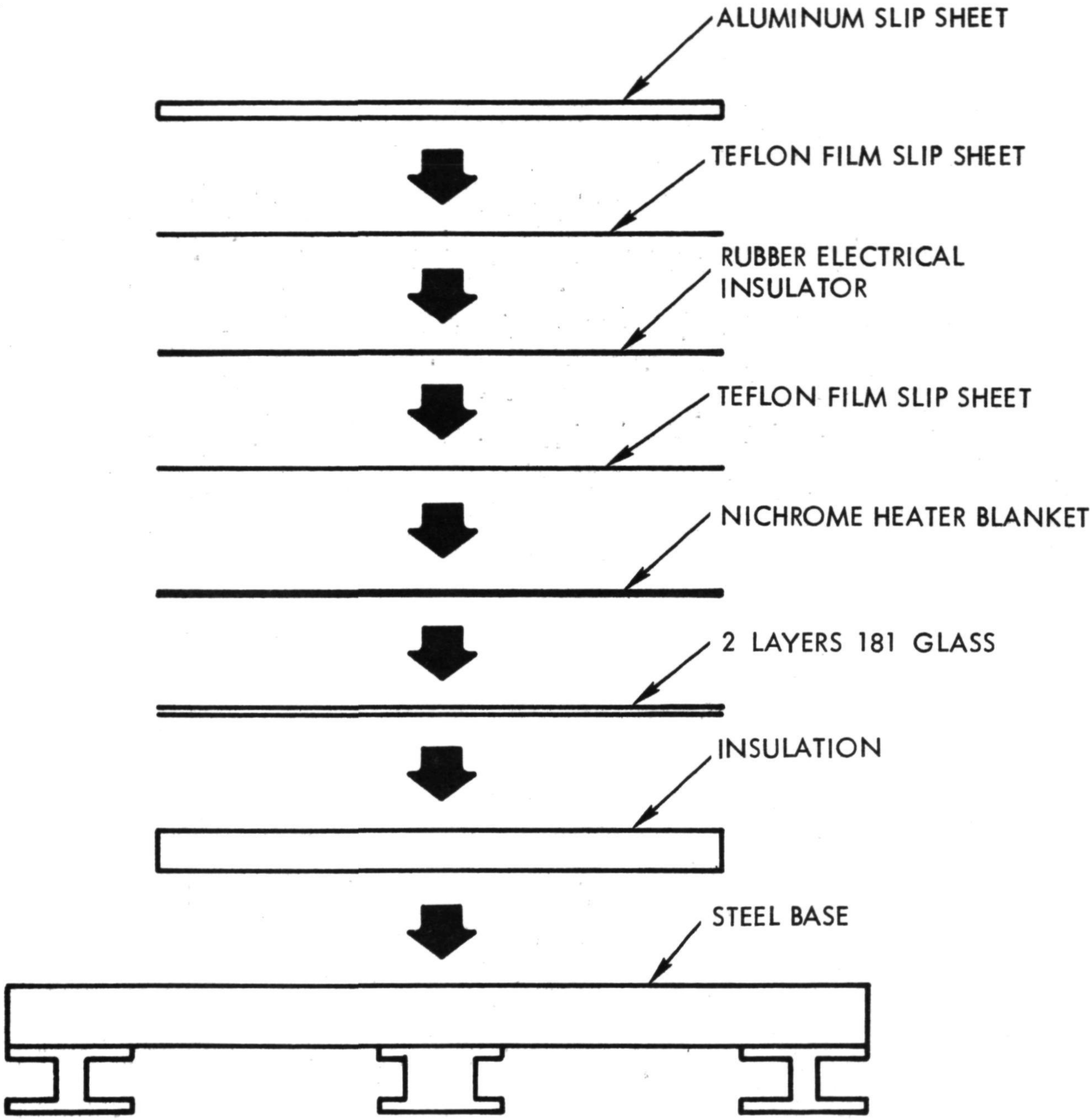


FIGURE 21.- COOL TOOL (CROSS-SECTION SKETCH)

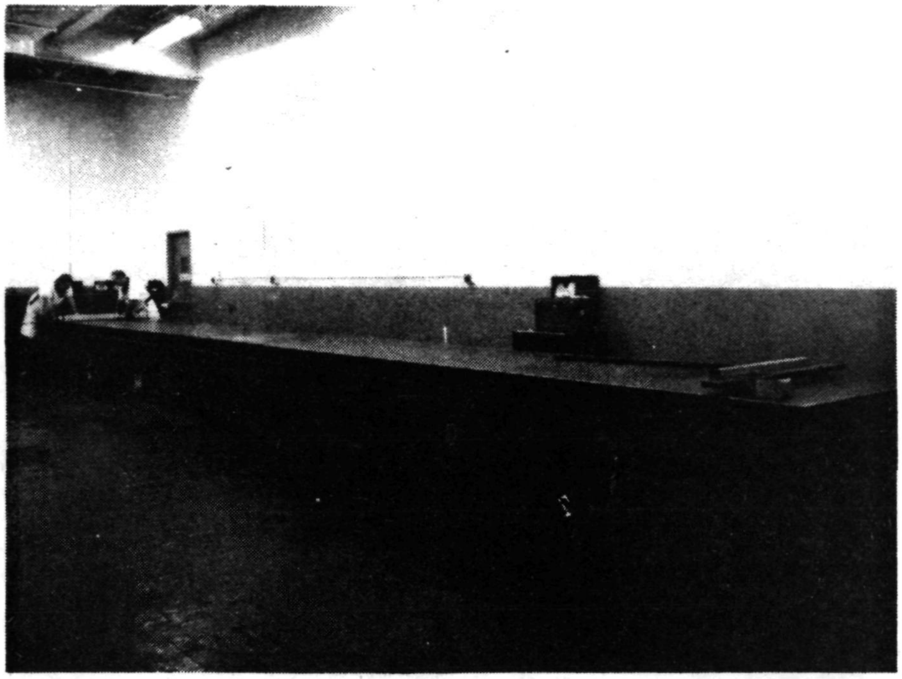


FIGURE 22.- COOL TOOL: FORTY FOOT TOOL FOR BONDING BORON-EPOXY STRIPS TO PLANKS AND STRINGERS.

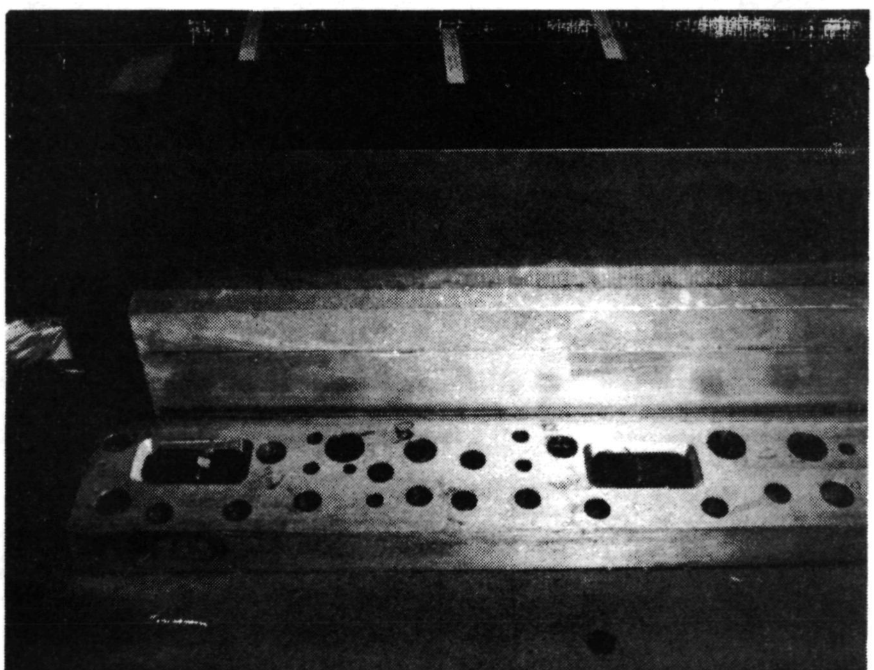


FIGURE 23.- STRENGTHENED END RESTRAINT OF COOL TOOL

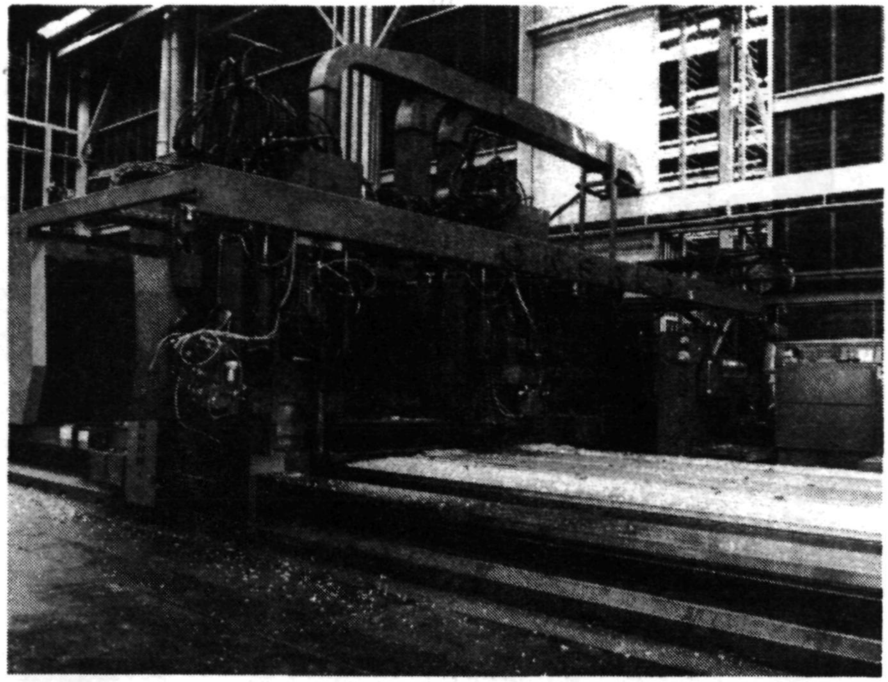


FIGURE 24. - PANEL MACHINING

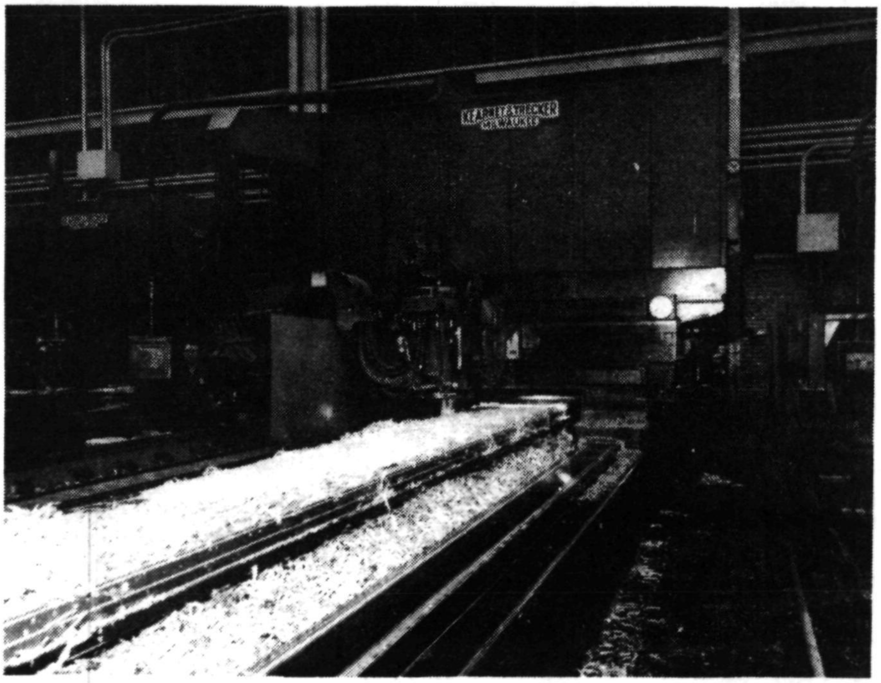


FIGURE 25. - PANELS BEING MACHINED ON NUMERICALLY-CONTROLLED MACHINE TOOLS

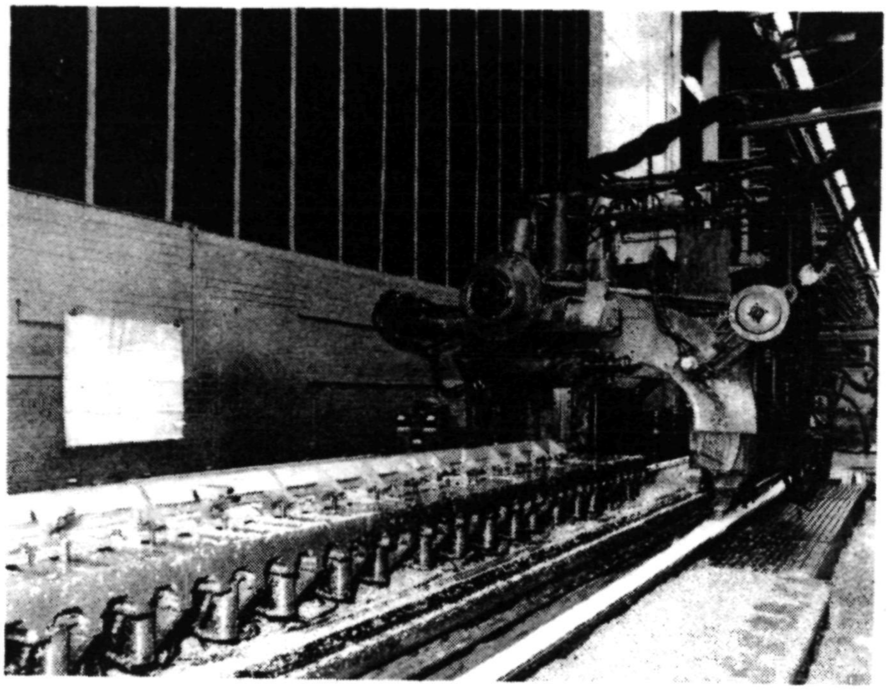


FIGURE 26.- STRINGER BEING MACHINED ON NUMERICALLY-CONTROLLED MACHINE TOOL

versus those machined. Extremely tight tolerances were imposed on panel thicknesses, and this created some relatively severe machining difficulties. The allowable deviation from nominal thickness was set at +0.038 cm, -0.012 cm (+0.015 in., - 0.005 inch). Most of the unacceptable panels resulted from an inability to hold the negative tolerance during machining of the very thin sections of the 11-meter (38-foot) long panels.

4.3.2 Shot Peening and Sulfuric Acid Anodizing

After machining, the panels and stringers were shot-peened and sulfuric acid anodized. The parts were shot peened with cut wire shot at a controlled velocity to produce residual compressive surface stresses on the peened surface. This process is standard and improves fatigue life, and, to some extent, increases resistance to stress corrosion.

After shot peening, the panels and stringers were sulfuric acid anodized over the entire surface (using normal production tanks) and spray-coated with chem-mill maskant to protect this anodic coating during subsequent chromic acid anodizing steps. The parts were then stored until needed for the laminate-to-metal parts bonding operation.

4.3.3 Preparation for Bonding

The panels and stringers were prepared for boron-epoxy laminate-to-metal bonding by removal of the chem-mill maskant from the bond surface areas. This was accomplished by lightly scribing the outline in the maskant and peeling the maskant from the part, as illustrated in Figure 27. The previously applied sulfuric acid anodize was removed from the unmasked surface and replaced with a reduced chromic acid anodize. These areas were primed for protection and to provide the proper surface for bonding. The balance of the maskant was then removed to complete the preparation.

4.4 REINFORCING LAMINATE FABRICATION

A total of 222 laminates were made during the fabrication program in 24 autoclave runs. This total was comprised of 110 skin laminates and 112 stringer laminates, and included thirty laminates which were used in tests, unusable due to tool discrepancies, damaged in bonding and/or metal machining after bond completion, and those scrapped for poor hole quality. One of the skin laminates was found to be inexplicably warped in the plane of the laminate and was replaced. Laminate replacements are tabulated in Table III. Only one laminate was scrapped due to direct laminate quality. Seven laminates, representing only 3.1% of all laminates fabricated, were replaced because of unacceptable fastener holes.

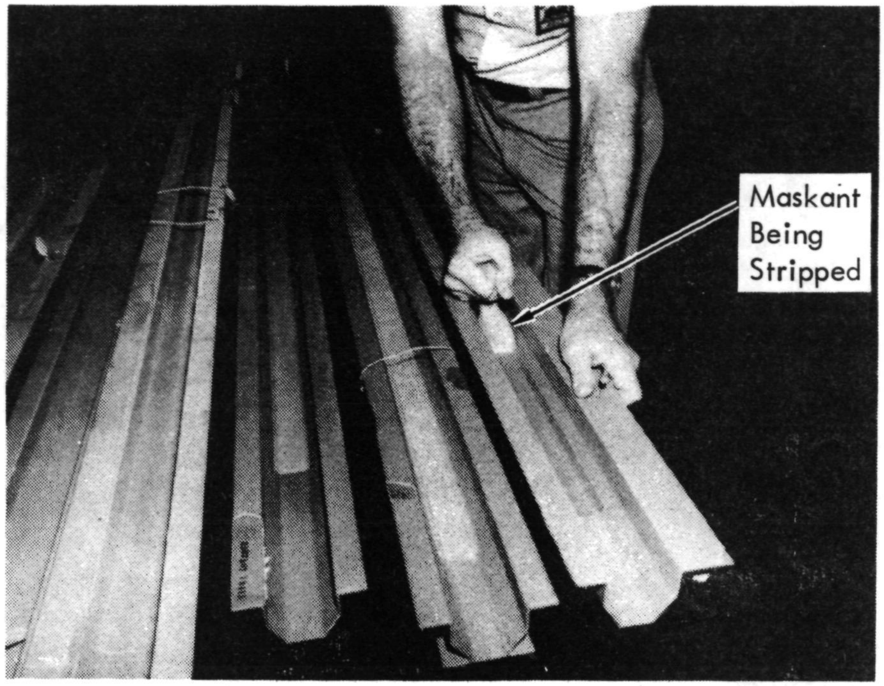


FIGURE 27.- STRIPPING CHEM-MILL MASKANT FROM STRINGER BOND AREA

TABLE III.- LAMINATE REPLACEMENT SUMMARY

Due to:	Skin Laminates	Stringer Laminates	Total
Laminate Tool Discrepancies	5	4	9
Bonding:			
Metal Crippled	4	4	8
Laminate Slipped	1	0	1
Laminate Verification Tests	3	0	3
Metal Machining After Bond	0	1	1
Unacceptable Holes	0	7	7
Warped Laminate	1	0	1
Total	14	16	30

The laminate layup and cure sequence was carefully planned to minimize the number of required autoclave runs and to obtain the maximum utilization of the curing dams. The autoclave tool accommodated a mixed load of four of each width dams in each load. Shorter laminates were made in combinations within a single dam, where possible, and some dams were used in every autoclave load. By such multi-purpose use, it was possible to make a ship's set of laminates (64 laminates, total) in only six autoclave runs with only 23 dams.

4.4.1 Laminate Layup

The boron-epoxy laminates were laid directly onto an Armalon peel-ply in the bottom of the appropriate dam, using the tape-dispensing tool. When this tool was rolled, by hand, along its track, the tape was automatically placed into the dam and compacted by a series of rollers attached to the dispenser, as shown in Figure 28. The dams were scribed with the locations of the beginning and ending of each ply where the boron-epoxy tape was cut with scissors.

The predrilled, cleaned, and primed titanium shims were coated with adhesive and located between the specified boron-epoxy plies. Fastener holes were punched through the ply buildup, as illustrated in Figure 29, at intervals determined by the number of plies between shims. The maximum number of plies punched in a single punching was seven. Pins were placed in some of the holes to prevent titanium shims and the boron-epoxy plies

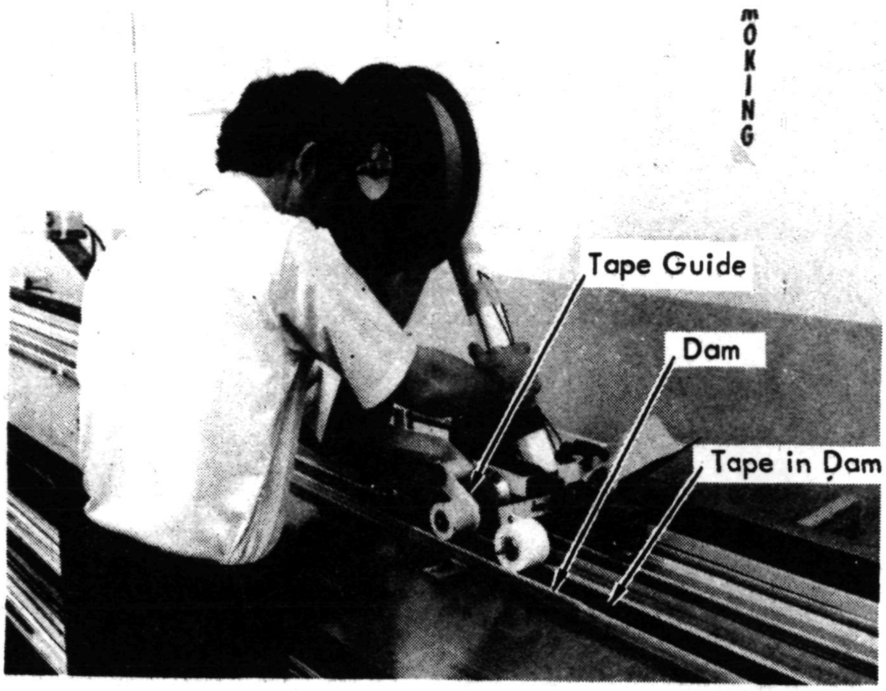


FIGURE 28.- LAMINATE LAY-UP

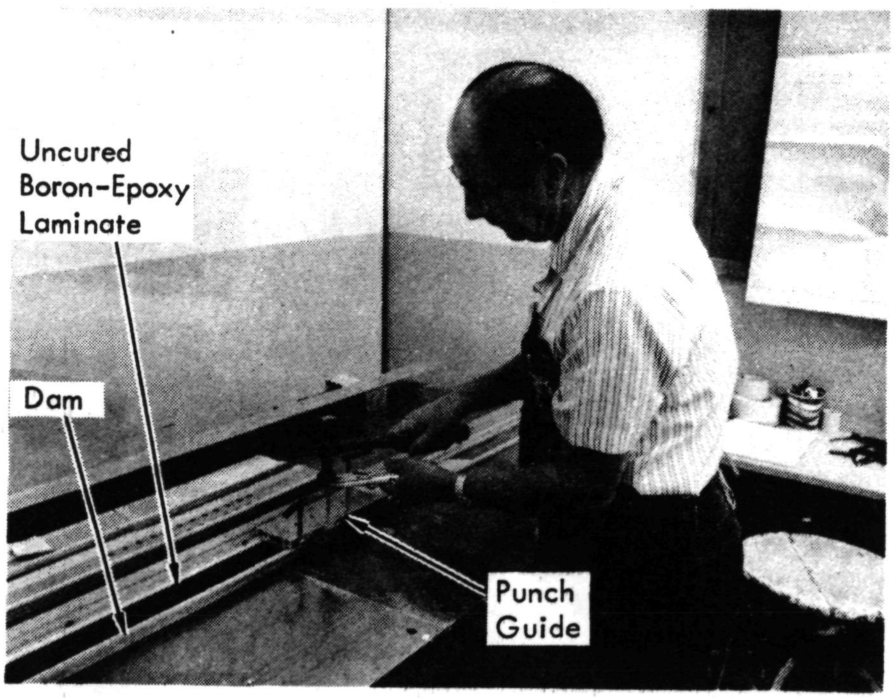


FIGURE 29.- PUNCHING FASTENER HOLES IN LAMINATE LAYUP

from shifting during subsequent ply lay-up. This activity is shown in Figure 30. The tape laying, punching, shimming, pinning sequence was repeated until all the plies for each specific laminate were in place. The holes were then plugged with teflon pins to prevent filling with resin during cure. One hole was plugged with a steel pin to better maintain hole alignment between the laminate and titanium shims.

Prior to laying the boron-epoxy tape into the dams, the dams were taped with teflon tape to prevent adhesion of the laminating resin to the dam. New taping was required for each use of a dam. Although this was a relatively slow process, attempts to use commercially available spray mold release agents were not effective.

After completion of the laminate layup, a layer of Armalon (teflon-coated fiberglass) was placed over the laminate and one ply of fiberglass bleeder cloth for each eight plies of laminate was added. A caul plate combination of 0.159 cm (1/16 inch)-thick rubber and 0.318 cm (1/8 inch)-thick aluminum, was wrapped in teflon release film and placed over the layup in the dam. The rubber side was placed next to the laminate to improve uniformity of pressure being applied to the tapered laminate. The loaded dam was then physically removed from the layup table and placed on the autoclave cure tool, as shown in Figure 31. When sufficient laminates were laid to fully load the tool, the dams were lined up against each other. Two layers of teflon film were placed over the whole assembly to prevent the resin from bleeding into the bagging materials which were added before cure. Figure 32 shows the autoclave tool with a full load of dams.

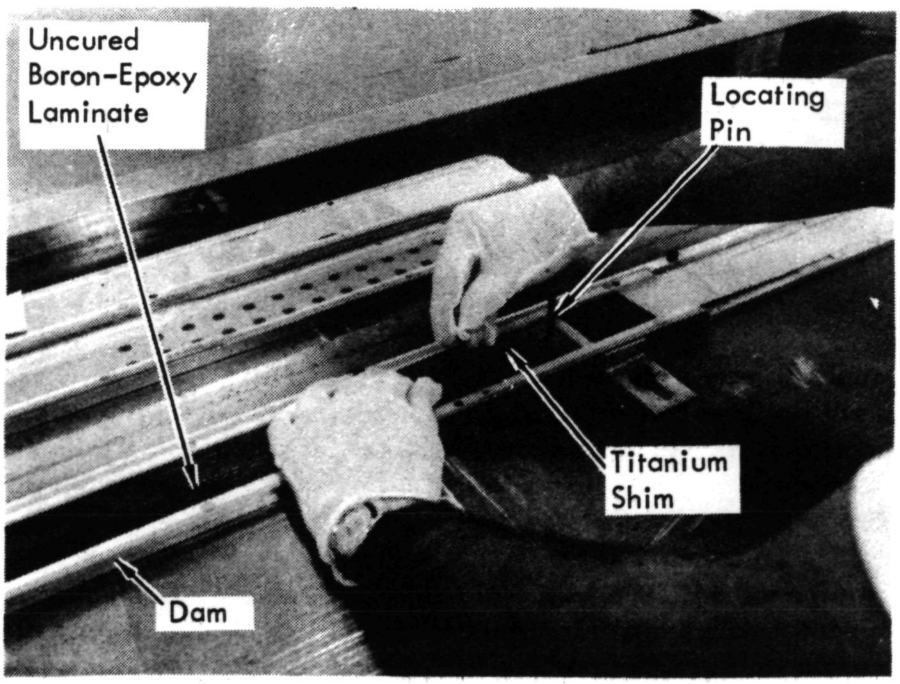


FIGURE 30.- PLACING PINS TO MAINTAIN LAMINATE-DOUBLER LOCATION

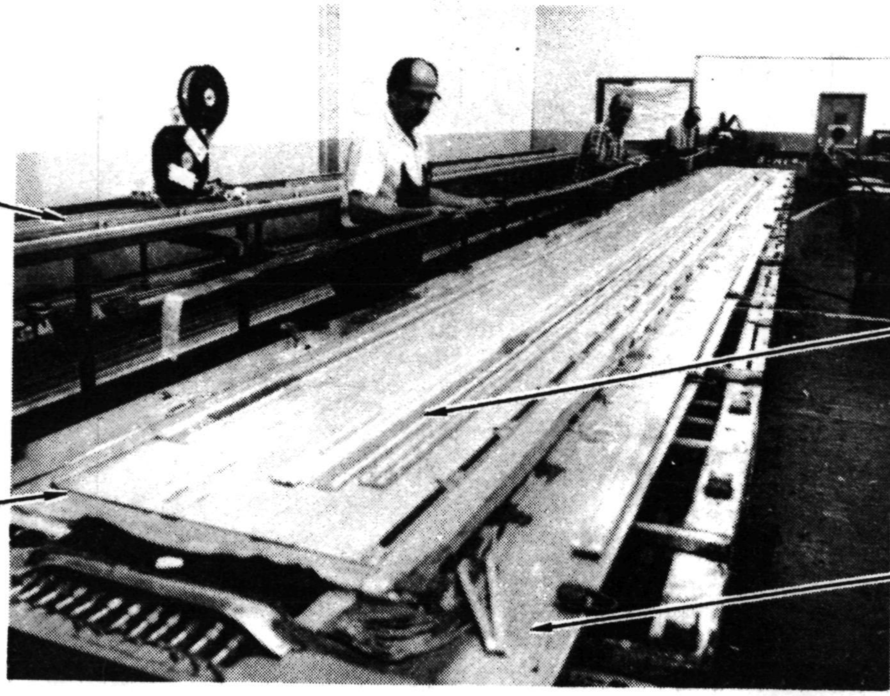


FIGURE 31.- TRANSFERRING LAMINATES IN DAMS TO AUTOCLAVE CURE TOOL

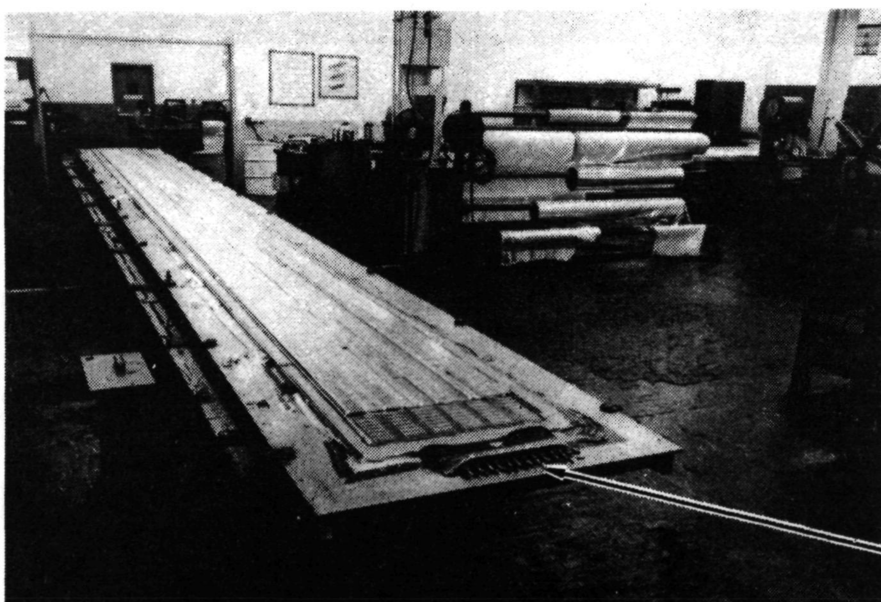


FIGURE 32.- LOADED AUTOCLAVE TOOL

Thermocouples were strategically placed on the layup to monitor cure temperature. Addition of cloth over the dams and nylon bagging film completed the total assembly. Required test specimens were included under the same bag. Figure 33 shows the completely bagged autoclave tool ready for transporting to the autoclave for cure.

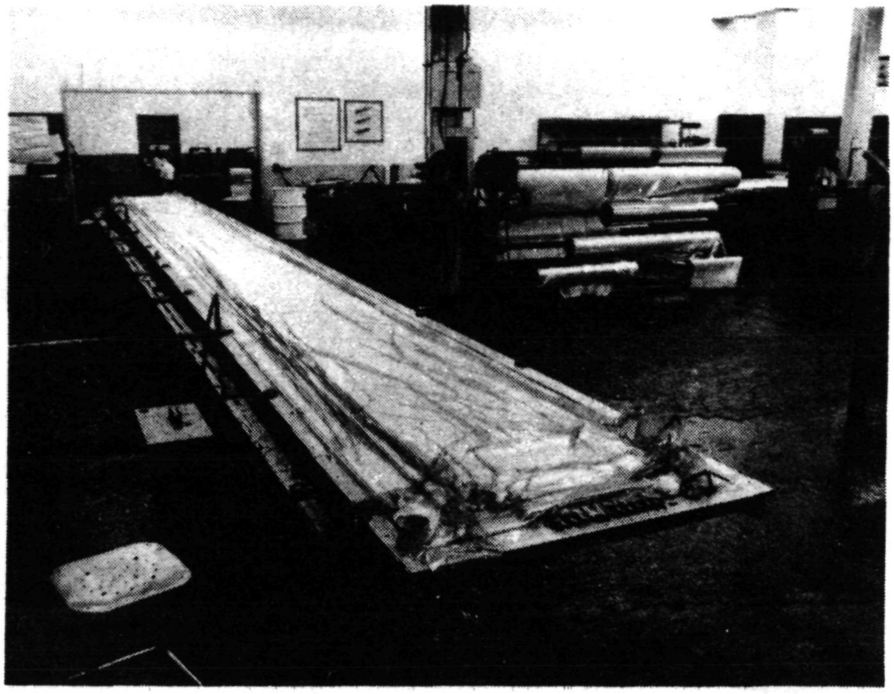


FIGURE 33. - BAGGED AUTOCLAVE LOAD

4.4.2 Laminate Cure

The bagged laminates, loaded on the cure tool, were subjected to full vacuum, and placed in the autoclave. The autoclave was then pressurized to 0.138 MN/m^2 (20 psi), and the vacuum vented to the atmosphere. The autoclave pressure was raised to 0.586 MN/m^2 (85 psi), where the temperature rise was started and maintained at a rate of 2.78 to 3.89°K (5 to 7°F) per minute until the final cure temperature of 450°K (350°F) was reached. This cure temperature was maintained for 90 minutes under full pressure. All temperature controls were then turned off, allowing the autoclave and tool to cool. The 0.586 MN/m^2 (85 psi) pressure was maintained until the temperature reached 339°K (150°F), at which time the autoclave was depressurized, the door opened, and the autoclave dolly removed. After debagging, each laminate was stamped with its individual assembly number, autoclave run number, and boron control number before removal from the dam.

This process was repeated until all laminates were fabricated. Throughout the laminate layup and cure, in-process inspection was performed and specific inspections were performed at the completion of each operation. Testing and evaluation of the process control specimens made with each laminate load is discussed in the Quality Control section of this report.

4.4.3 Preparing laminates for Bonding

After cure, the laminates needed only slight cleanup. Resin flash was removed by lightly sanding the sides of the laminates, as shown in Figure 34. After the flash was removed, the resin which had flowed into the prepunched, predrilled holes, was removed by drilling. The holes were then reamed to full size by running a diamond coated reamer through the holes, as shown in Figure 35.

The last operation, performed on the boron-epoxy laminates before being staged for bonding, was a complete ultrasonic inspection to ensure that there were no voids or delaminations in the ply buildup which could affect the integrity of the final product. Figure 36 shows the ultrasonic laminate inspection in progress.

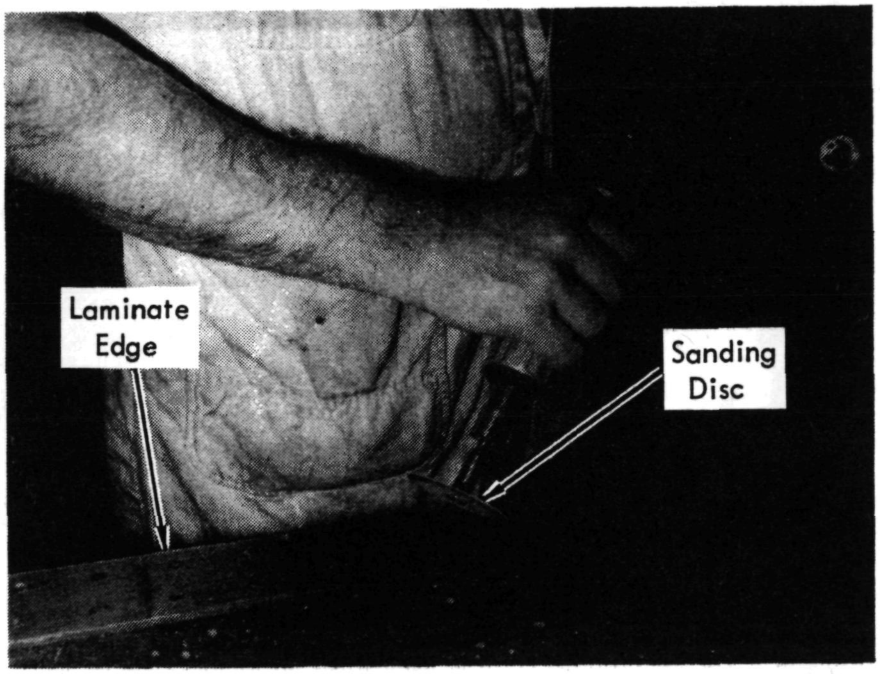


FIGURE 34. - LAMINATE CLEAN UP

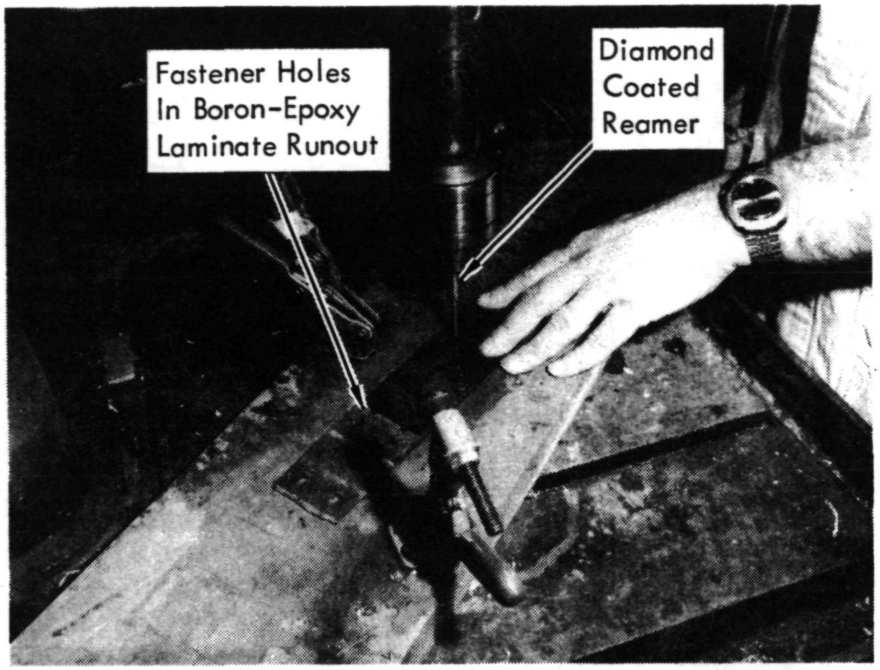


FIGURE 35.- REAMING LAMINATE HOLES

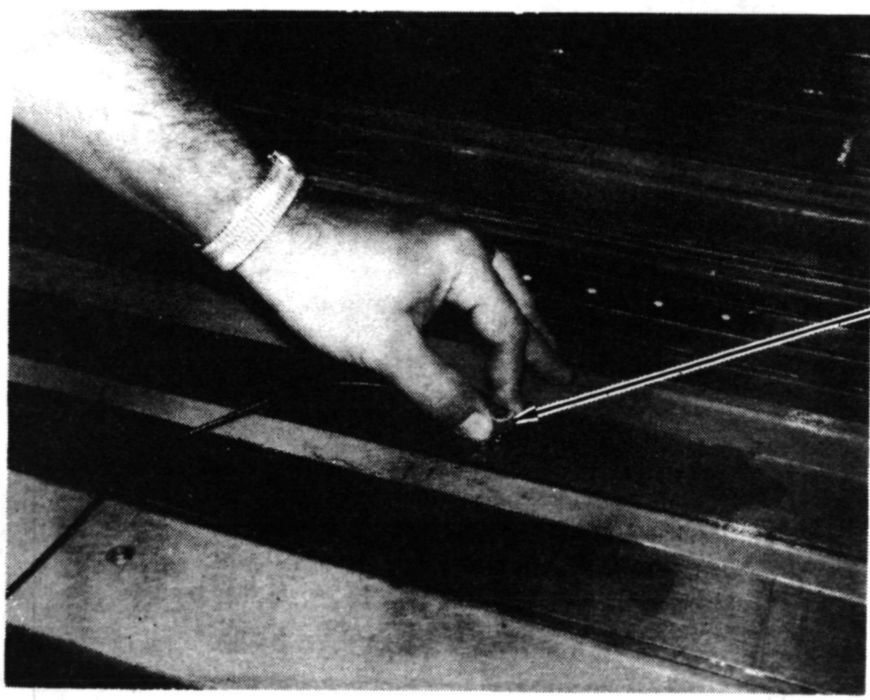


FIGURE 36.- ULTRASONIC INSPECTION OF REINFORCING LAMINATE

4.5 LAMINATE-TO-METAL PARTS BONDING

The boron-epoxy laminates were bonded to the planks and stringers on the "cool tool" described previously in this report. This tool incorporated integral heating. Pressure was applied to the laminate during cure by placing inflatable rubber/fabric tubes over each laminate to be bonded. Because of the large difference in thermal expansion of the two adherends, the aluminum was restrained during cure so that residual stresses were kept to a minimum at room temperature.

4.5.1 Bonding Laminates-to-Planks

Several of the plank sections were severely contoured on the tool side requiring complex shimming to allow the planes to lay flat against the tool surface and to provide intimate contact with the tool during cure. Shims ranging in thickness from 0.03 to 0.64 cm (0.012 to 0.25 inch) were required to achieve the required flatness. The most complicated panel required over 200 shims. Figure 37 shows the shims being taped in place on the outer surface of a wing plank.

The panels were placed in the "cool tool" after completion of the shimming operation, and the end restraint slide bar was adjusted to obtain a tight fit. On some panels, with ends as thin as 0.178 cm (0.070 in.), a doubler was pinned to the panel to prevent panel buckling in that thin area.

FIGURE 37. - SHIMMING OUTER SURFACE OF WING PLANK FOR BONDING

While the panel was being prepared and placed in the "cool tool," the laminate peel-ply was stripped and adhesive was applied to the bonding side of the laminates. The laminates were then placed in their respective locations and secured with teflon tape. Spacers were placed between laminates to prevent movement during bonding.

Pressure was supplied by fire hoses contained within aluminum channels. These pressure devices were placed over each laminate so that uniform pressure could be applied to the laminate. Steel bars were placed longitudinally along the tops of the aluminum channels to provide extra stiffness. In addition, longitudinal steel bars were placed on each side of the channels and throughout the entire panel length to prevent the panel from bowing away from the tool. Steel tie-down bars were placed across the entire system and bolted to the tool. A sketch of this system is shown in Figure 38.

Foam insulation was placed in the spaces between bars to hold the heat within the assembly. Figure 39 shows a closeup of the steel tiedown bars and the insulation.

The fire hoses were pressurized to 0.241 MN/m^2 (35 PSIG) using the plant air supply. The eight heater blankets were turned on, and by using a controller for each zone, the temperature was raised to $386 \pm 8.33^{\circ}K$ ($235 \pm 15^{\circ}F$) at a rate of $2.78 - 3.89^{\circ}K$ ($5 - 7^{\circ}F$) per minute. This temperature was held for ninety minutes to cure the adhesive. After cooling, the tool was disassembled and the next part was loaded. The set up and cure process was repeated until all laminates were bonded to planks. Figure 40 shows the tool during a bonding cure cycle.

After each bond cycle, the bonded areas were ultrasonically inspected for possible disruptions in the bond line. A number of minor edge voids and some under-laminate disbonds were recorded. Edge voids were generally repaired with EA 9309.1 room-temperature-cure adhesive. More extensive disbonds were repaired with combinations of adhesive, splice plates, and fasteners, as discussed in detail in another section of this report.

After the panel bond had passed inspection, holes were drilled through the metal plank using the predrilled holes in the boron laminate as a guide. Fasteners were then installed in the holes. Figure 41 shows the teflon plugs used during bond cycles to keep the predrilled holes from filling with adhesive. These plugs were easily removed, leaving a clean hole for final reaming.

Excess adhesive (which had been squeezed out at the sides of the laminates, and which could cause interference in later assembly) was removed and a protective edge-sealant was applied along the length of the laminate. Figure 42 shows the environmental sealant being applied.

After the sealant was applied, end cuts and door openings were machined, the panels were painted and stored until needed for box assembly. A typical completed panel is shown in Figure 43.

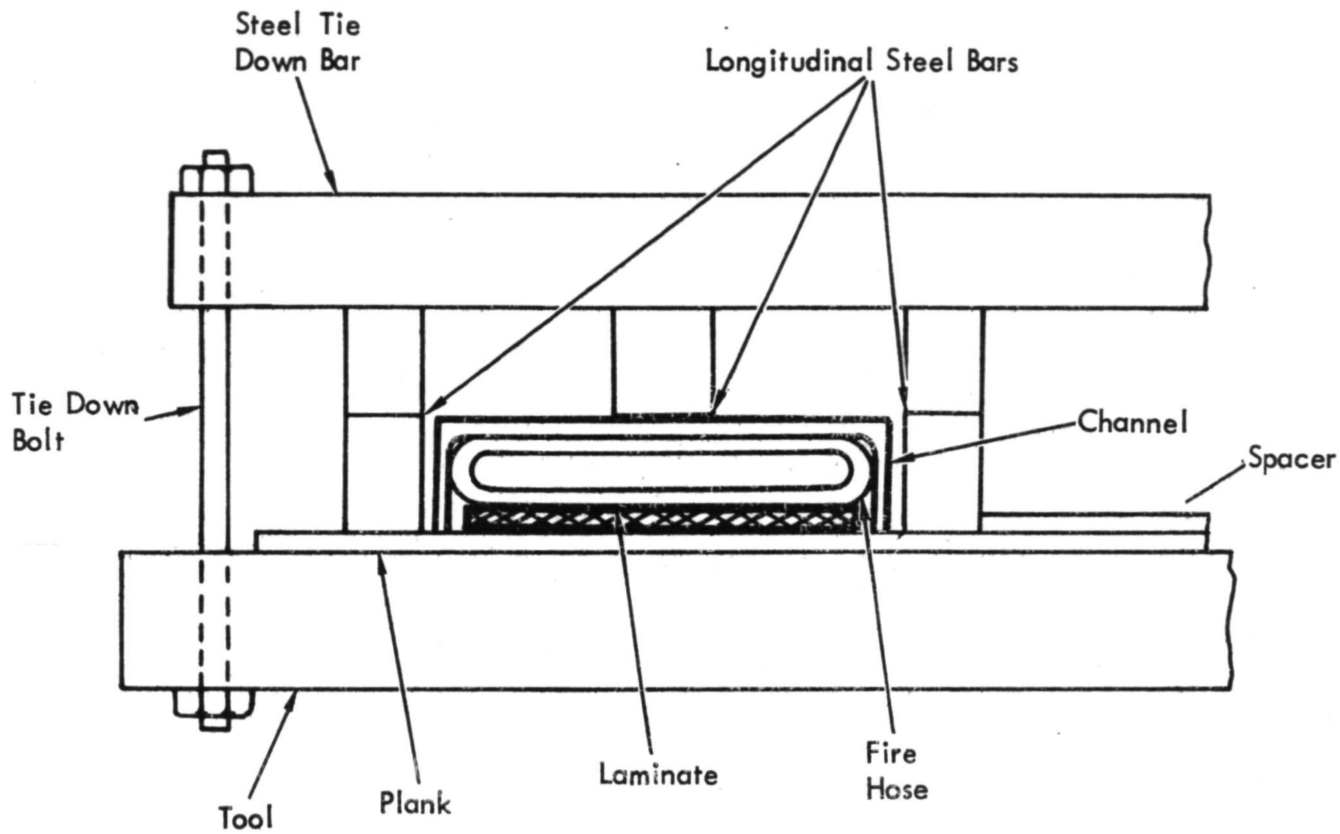


FIGURE 38.- BONDING FIXTURE

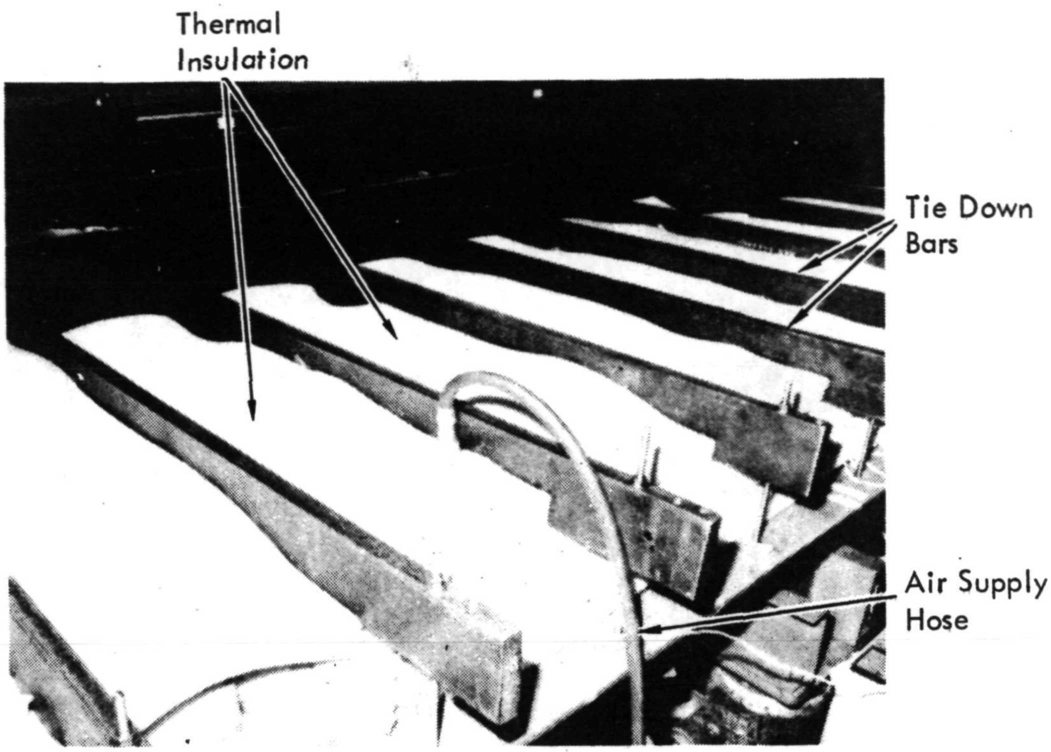


FIGURE 39. - CLOSEUP OF STEEL TIE DOWN BARS IN COOL TOOL

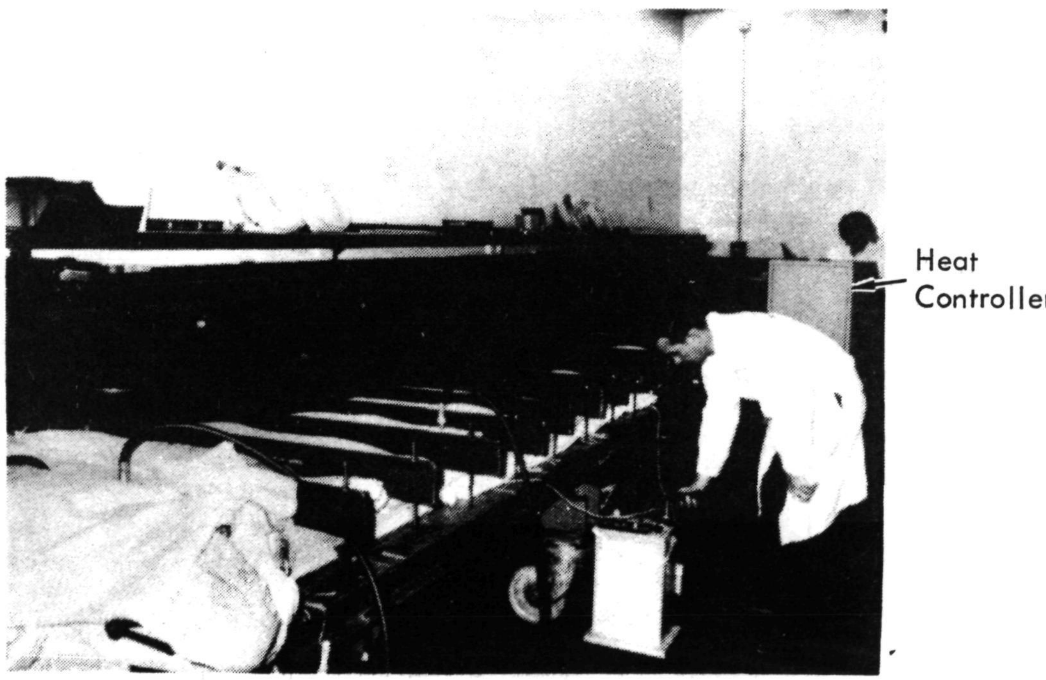


FIGURE 40. - COOL TOOL IN OPERATION

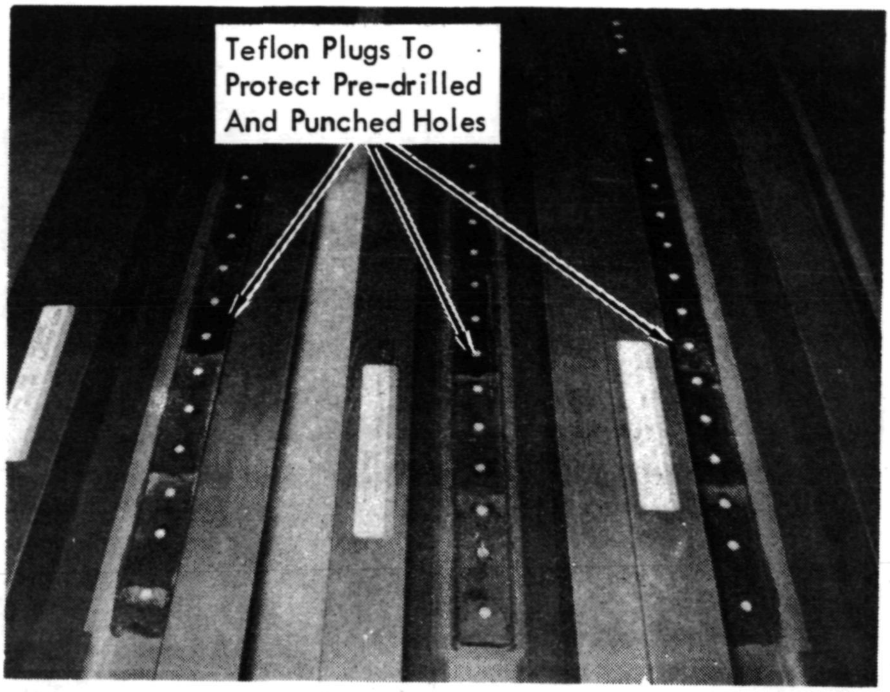


FIGURE 41.- LAMINATE RUN-OUT AREAS IN HAT-SECTION STRINGERS

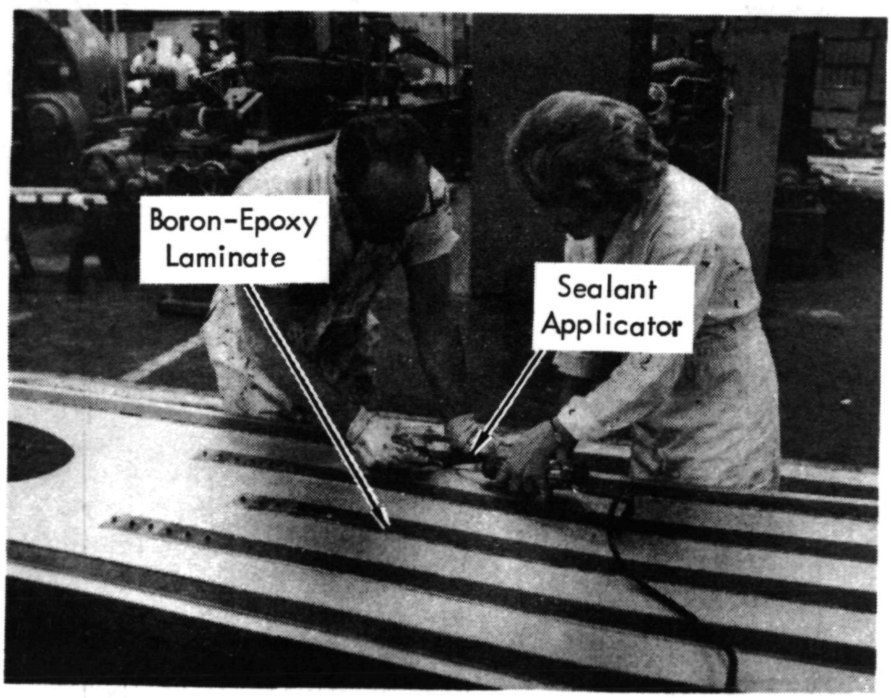


FIGURE 42.- APPLICATION OF ENVIRONMENTAL SEALANT TO LAMINATE EDGES

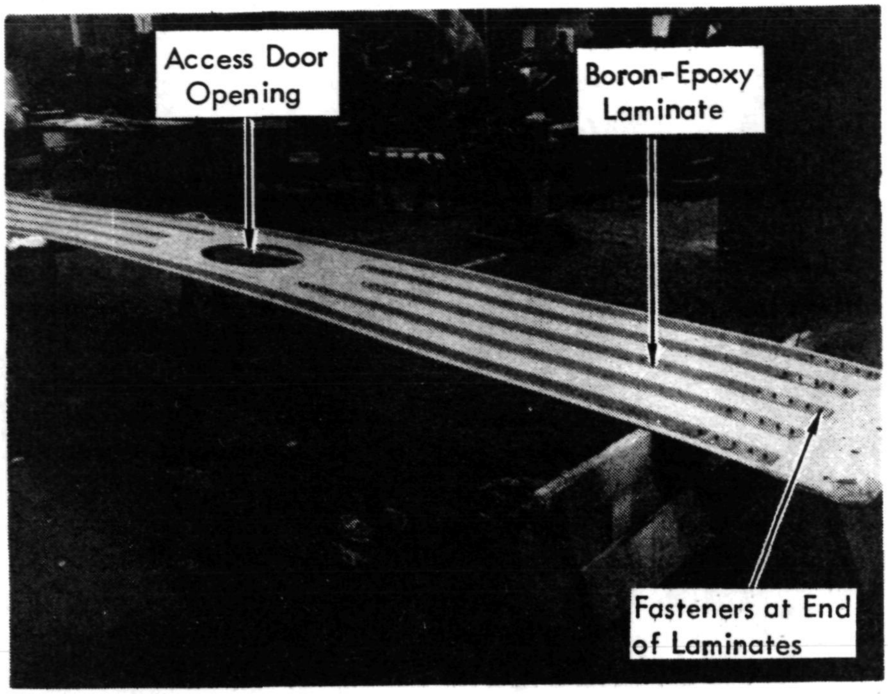


FIGURE 43.- COMPLETED LOWER SURFACE WING PANEL

4.5.2 Bonding Laminates-to-Stringers

The boron-epoxy laminate-to-stringer bonding was simpler than the laminate-to-panel bonding. The stringers did not require shimming. Also, they provided an integral containment for the pressurizing fire hoses, eliminating the channel and part of the large number of steel supports required by the panels.

In the box, some stringers are interrupted for doors. In the cool-tool process, all stringers were machined to the full box length. For those stringers which are interrupted, several laminates were bonded into the single stringer, which was later cut to provide the required lengths. Five stringer extrusions were simultaneously placed in the "cool tool" with the crown of the hat section against the tool, for each bond run. The appropriate boron-epoxy laminate, with a layer of adhesive on the bond side, was positioned inside each stringer. The laminates were taped along the edges with teflon tape to prevent movement during the bond cycle. The fire hose pressure vessel was placed inside the stringer on top of the laminate and an aluminum retainer plate to contain the fire hose was placed over the pressure vessel as shown in Figures 44 and 45.

The assembly was insulated and the steel cross bars were installed in the same manner as used for the laminate-to-panel bond. The laminate-to-stringer bond cure cycle was identical to that previously described for the laminate-to-panel bond cure cycle. Figure 46 shows a load of five stringers being unloaded. The stringer bonds were ultrasonically inspected for voids and delaminations, as shown in Figure 47. Only one small disbond was documented in inspections of 72 stringers.

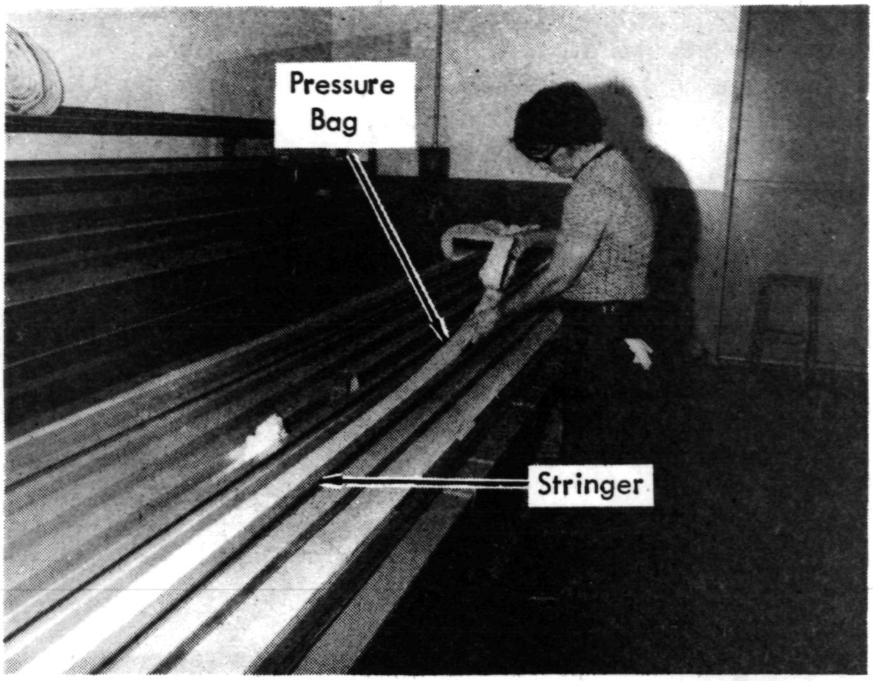


FIGURE 44.- INSTALLATION OF PRESSURIZING HOSE IN STRINGER

FIGURE 45.- INSTALLATION OF PRESSURE RETAINING PLATE

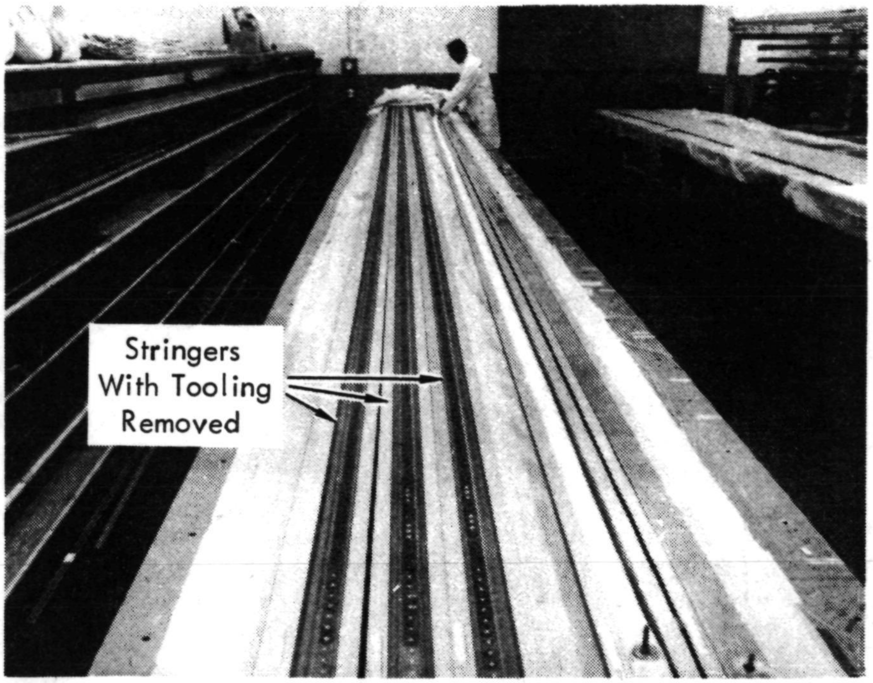


FIGURE 46. - LAMINATE-TO-STRINGER BOND LOAD BEING UNLOADED

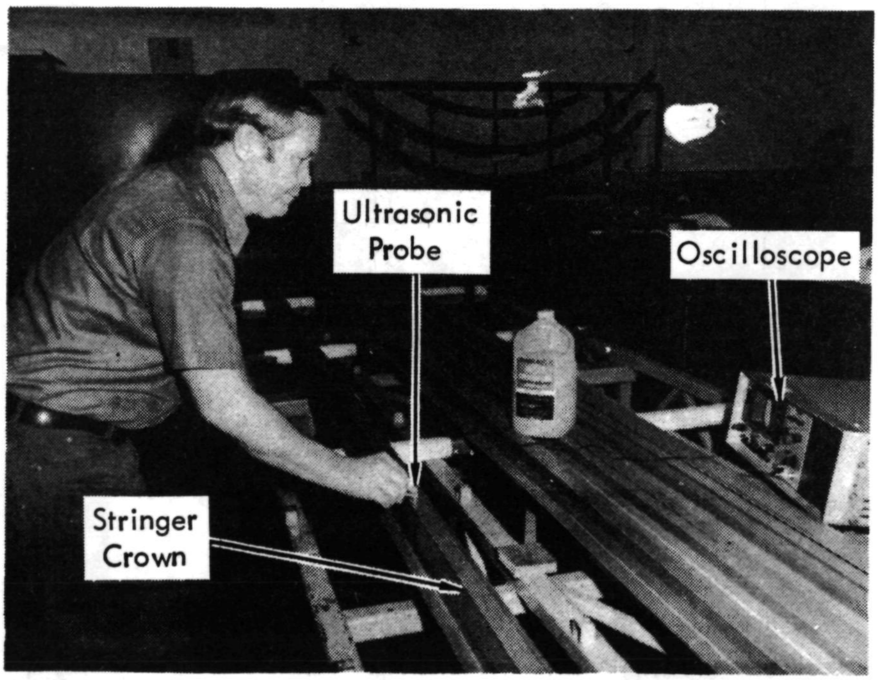


FIGURE 47. - ULTRASONIC INSPECTION OF STRINGER/LAMINATE BOND

At the laminate run-out areas, holes were drilled from the laminate side using drill blocks as shown in Figure 48. The stringers were then cut to the proper length and the ends were machined to the standard C-130 configuration. In this configuration, the stringer ends are tapered to mate with the rainbow fitting and to provide a better load transfer path. Figure 49 shows the tapered cut being made at the end of a stringer.

Wing box assembly sequences require, on some stringers, that blind fasteners be installed on the assembly line. All other fasteners were pre-installed. The holes, which were to receive blind fasteners had an aluminum doubler bonded to the laminate to prevent the boron-epoxy from being fractured during fastener installation. These doublers were bonded using a room temperature curing adhesive. Figure 50 shows the doublers in place.

After environmental sealant was applied to the edges of the bondlines, stringers were painted, and stored until needed for wing box assembly.

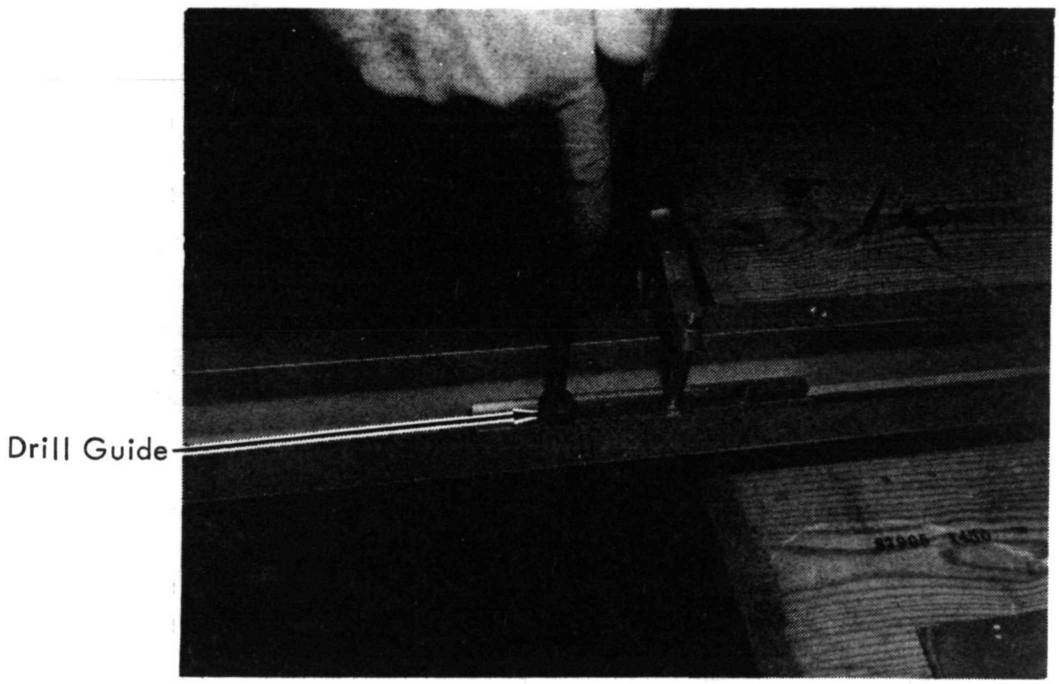


FIGURE 48. - DRILLING HOLES AT LAMINATE END

4.6 WING BOX ASSEMBLY

In the C-130 production sequence, the center wing is assembled as an individual component and delivered to the production line for mating with the center fuselage components as the first operation in total aircraft assembly. The composite-reinforced center wing box was assembled in the standard sequence, using the same jigs, fixtures, and personnel. Minor differences due to relocation of fasteners or brackets were accommodated within the existing assembly stages with only slight procedural modifications to account for the presence of the boron-epoxy reinforcements.

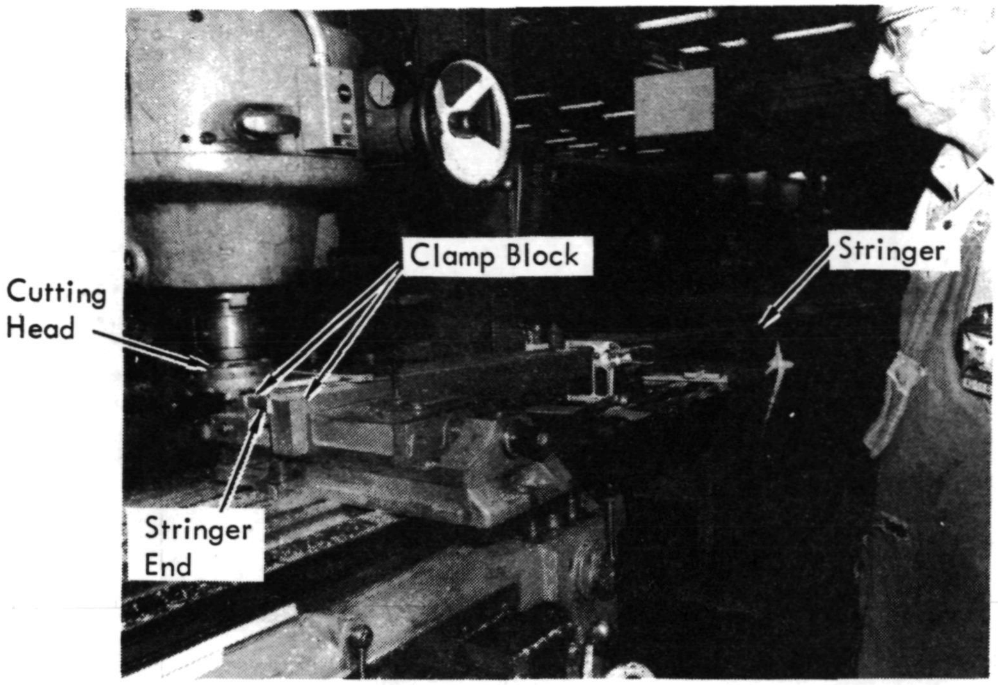


FIGURE 49. - STRINGER END MACHINING

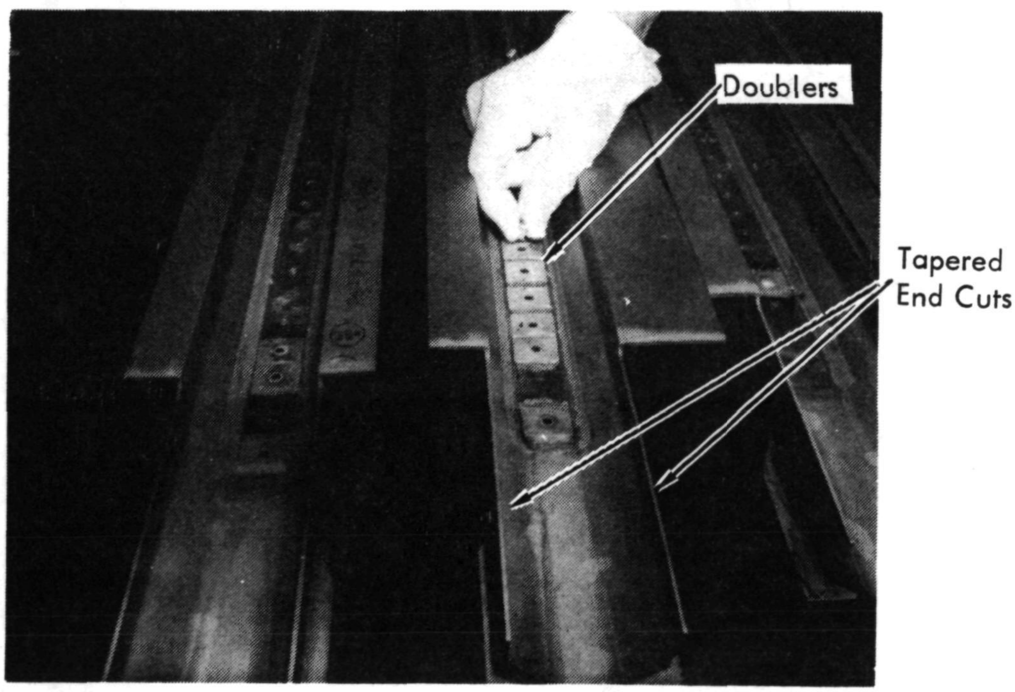


FIGURE 50. - ALUMINUM DOUBLERS IN STRINGERS

When delivered to the center wing assembly area, the planks and stringers were ready for use. All holes common to the reinforcing laminate and metal detail were pre-drilled and reamed before delivery. Assembly of the center wing box began with a "stringer build-up" step, illustrated in Figure 51, where the stringer-to-rainbow fitting straps were installed. Miscellaneous clips and brackets were also installed on the stringers in this step.

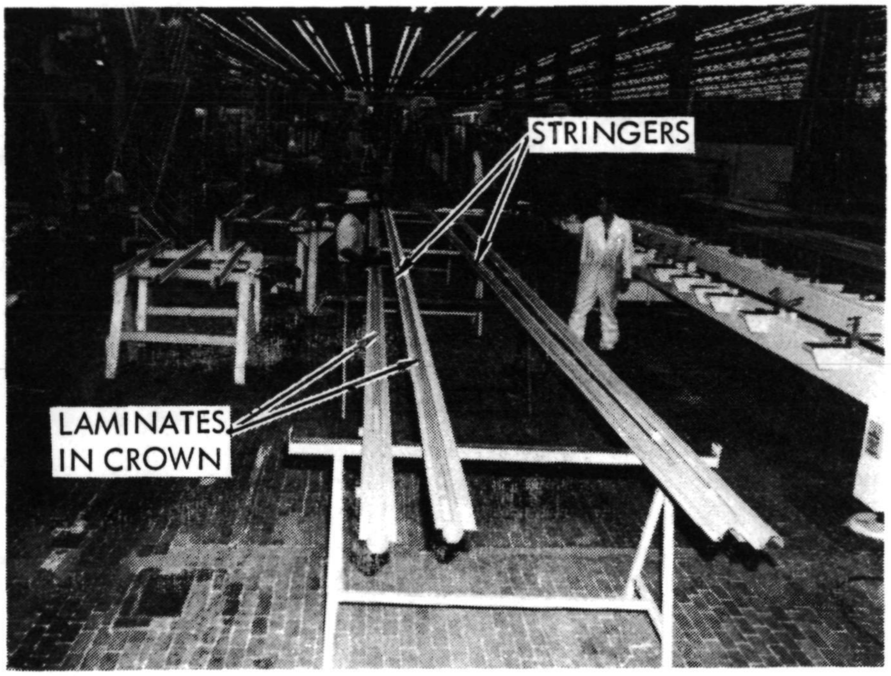


FIGURE 51. - STRINGER BUILD-UP

The upper and lower wing surfaces were assembled individually in first-stage jigs. The stringers were loaded into the jig and mated with the W.S. 220 joint (rainbow) fittings. The skin panels were then fitted into the jig, fastened to the rainbow fitting and "tacked" to the stringers at numerous points to maintain locations during subsequent automatic riveting. Also, in this stage, some of the door doublers and plates were installed.

Figure 52 shows the test article upper surface inthe first-stage jig, at the time when planks were being fitted. The lower surface is shown in Figure 53 after completion of the first-stage assembly. Pickup bars were installed for handling and moving the surfaces during assembly.

A large number of the fasteners in both the upper and lower surfaces were installed by automatic machinery. The GEMCO Drivmatics used are tape-controlled machines which automatically drill, countersink, and wet install fasteners. In some surface areas, where fasteners were different from standard, the fasteners were installed by hand.

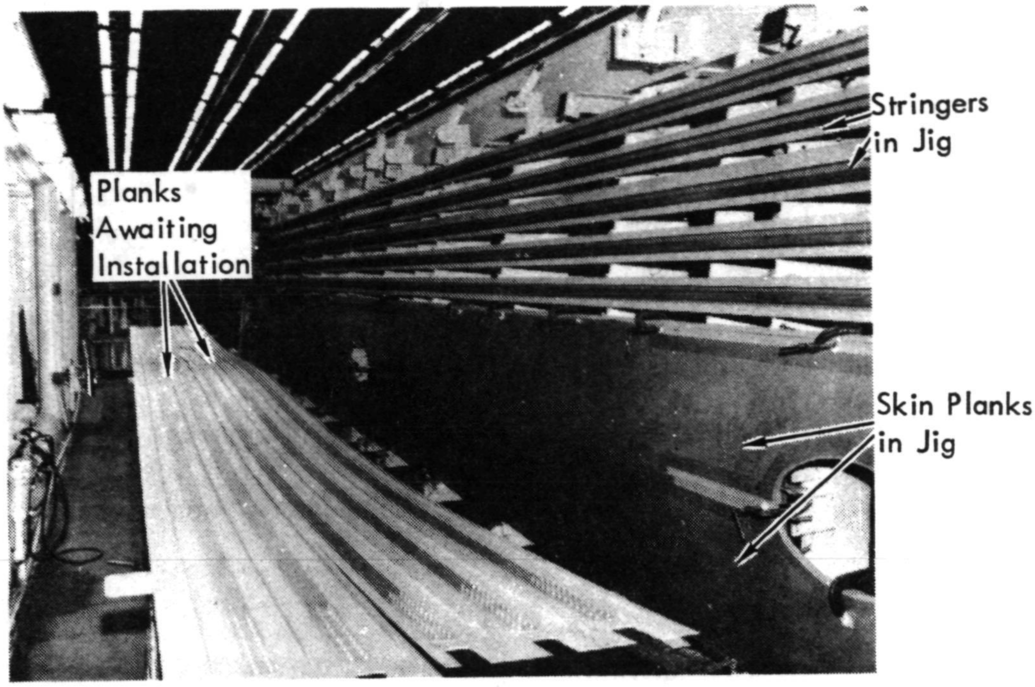


FIGURE 52.- UPPER SURFACE IN FIRST-STAGE ASSEMBLY FIXTURE

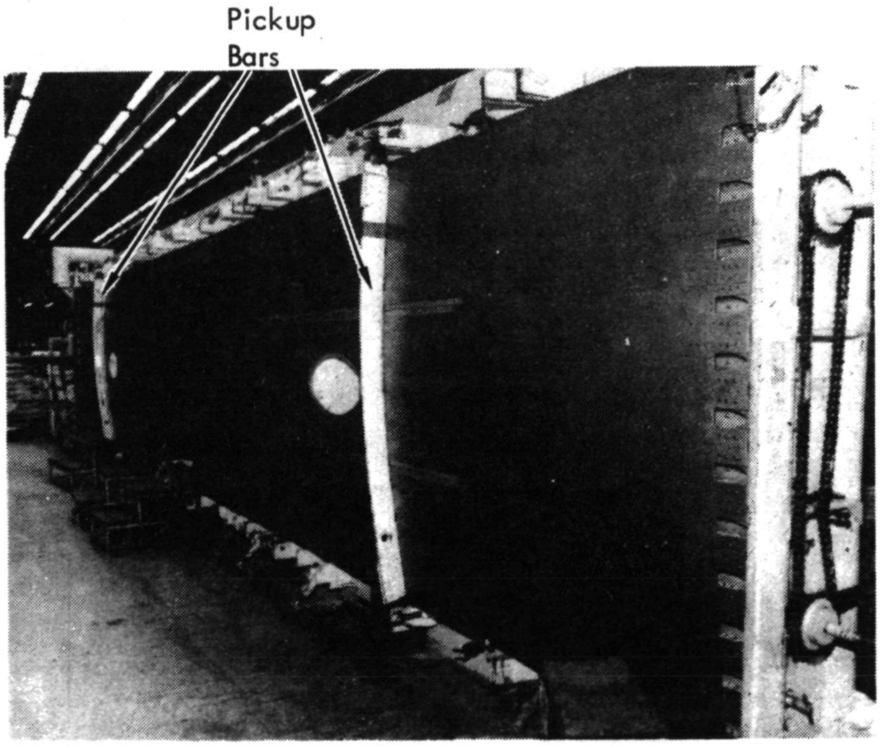


FIGURE 53.- LOWER SURFACE IN FIRST-STAGE ASSEMBLY FIXTURE

Following the Drivmatic operation, shown in Figure 54, the upper and lower wing surface assemblies were moved into second-stage jigs for "pick-up" operations. In these jigs, certain other detail parts were installed in preparation for mating in the center wing box mating jig. These details include rib caps, spanwise panel splices, and miscellaneous hardware.

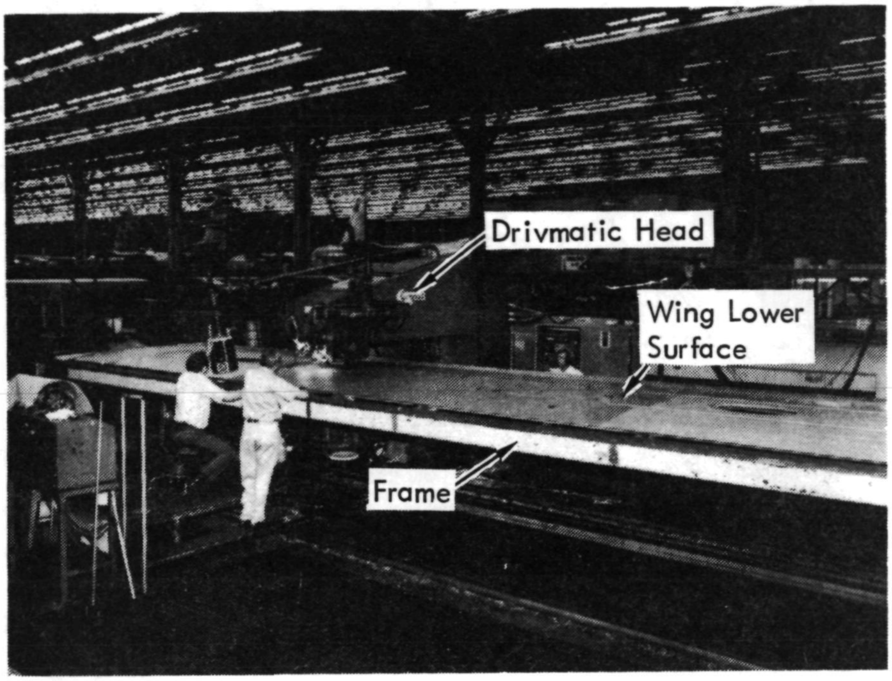


FIGURE 54. - WING SURFACE IN DRIVMATIC

Figure 55 shows the static test article being assembled in the box beam fixture. In this jig, the upper and lower surfaces were mated and assembled with the front and rear beams along with all the bulkheads and braces.

The engine mounts and front beam were loaded into the fixture first. The surfaces were then located in place by using the rainbow fittings as locators, and the pre-assembled rear beam-trailing edge unit was installed. The box is structurally completed in this position with rib webs and fuselage attach angles being installed.

After completion of the operations in the center wing box jig, the completed box was moved to the final assembly position where parts such as engine fairings and leading edge formers were installed. After assembly, the box was sealed and painted in accordance with standard C-130 production processes. It was then ready for either test or installation, as designated.

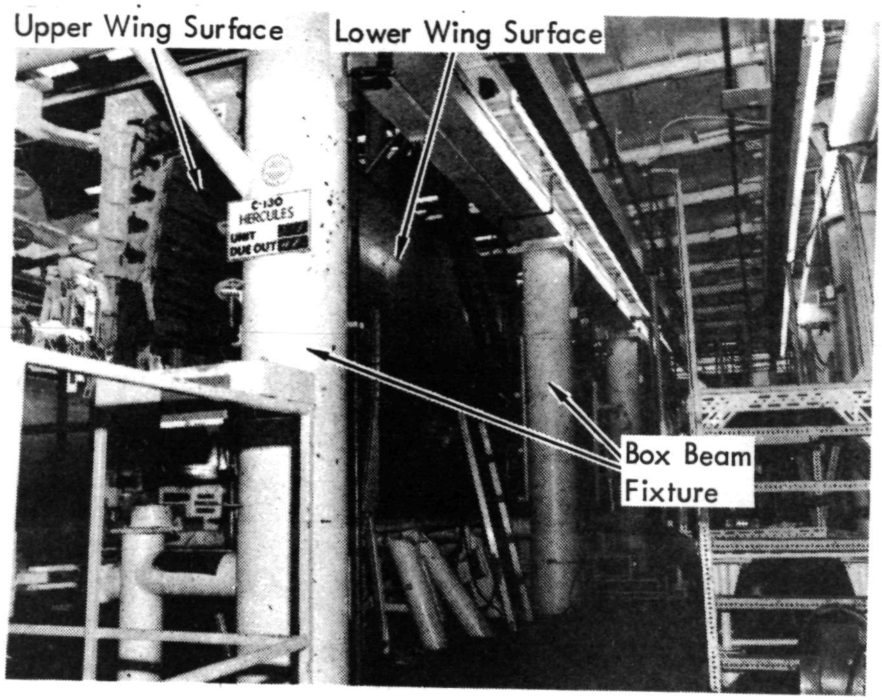


FIGURE 55.- TEST ARTICLE IN BOX BEAM FIXTURE

5.0 RELIABILITY AND QUALITY ASSURANCE

A reliability and quality assurance program was continued in accordance with the NASA-approved program plan. The plan, which complies with required elements of NASA specifications NHB 5300.4 (IA and IB), was updated during Phase II to incorporate program changes in the subsequent production and test phases.

5.1 RELIABILITY PROGRAM

The objective of the C-130 composite-reinforced wing box reliability program is to attain a high level of inherent reliability in system design; to assure that this level of reliability is not degraded throughout the production, test, and operational phases; and to provide to NASA the assurance and visibility that specified reliability requirements are achieved. The reliability objective during Phase III was to ensure a high degree of hardware conformance to detail design requirements and thereby minimize any degradation from the inherent design reliability level. Reliability achievement was dependent primarily upon manufacturing activities including tooling, planning, scheduling, fabrication, assembly, and inspection. Program status review and monitoring techniques developed and effectively applied in Phases I and II were continued during Phase III. Bi-weekly program status review meetings with the program manager and key program participants was an important tool for bringing problems into focus and assuring timely corrective action. These meetings were accelerated to a weekly frequency during more critical periods in the program.

Numerous problems were encountered, which is typical with new production operations. These problems, which are summarized elsewhere in this report, encompassed material properties, tooling, instrumentation, materials availability, workmanship, and design interference. Problem solutions and discrepancy dispositions were developed and implemented with a strong awareness and motivation toward program reliability objectives. Additional tests were run to determine and confirm major problem causes. Although there were some defects in laminate holes and metal-to-laminate bond lines, no one defect is considered a serious threat to static or fatigue strength margins. Combinations of defects are more difficult to assess and the fatigue test program will be carefully monitored to determine if an impact occurs. A marked reduction in the number of discrepancies on successive wing box hardware shows significant learning and raises the confidence level assessment of the flight articles over the static/fatigue test article. Although there were several aluminum workmanship defects, the level of inspection employed on this program suggests an overall quality level which equals or exceeds that of a production all aluminum wing box.

Another significant factor contributing to the overall quality level was the decision to halt production and revise the program schedule when material shortages and adhesive problems were encountered. The revised schedule permitted a more thorough analysis of the

adhesive flow problem, the selection of an alternative adhesive, and the formulation of a test plan to ascertain the acceptability of laminates already produced. A more reliable composite-reinforced center wing box was the result.

In retrospect, the Phase III experience strengthens the conviction that reliable, high quality level composite reinforced aluminum structures can be produced using elevated temperature curing adhesives and the "cool tool" restraint system. Successful implementation requiring defined process controls, sufficient inspection check points, and adequately trained personnel was clearly demonstrated.

5.2 QUALITY ASSURANCE PROGRAM

Quality assurance activities were very extensive during Phase III to assure that the three C-130 reinforced center wing boxes retained the same high quality as that experienced with the production aircraft. These activities included acceptance inspection for all incoming raw materials, and extensive in-process inspection of all manufacturing steps, including step-by-step sign-off for satisfactory completion of preselected operations. In-process inspection included dimensional inspection to assure that the parts were in conformance with the applicable drawing. Tests were conducted on process control to verify the quality of the boron-epoxy laminates and of the laminate-to-metal bonds.

Mandatory characteristics were defined and identified on appropriate drawings. The mandatory characteristics were those qualities requiring 100 percent inspection and encompassed those qualities associated with fabrication of the boron-epoxy laminate assemblies and the bonding of these assemblies to the aluminum skins and stringers. In both cases, the boron-epoxy laminate and its bond to the aluminum details, ultrasonic inspection was conducted to verify acceptable quality. The inspection of both lamination and bond was conducted in addition to normal process control specimen testing.

Ultrasonic inspection techniques used are those previously reported in NASA CR-112272, Reference 2. No significant voids occurred in stringer/laminate bonds, nor did any of the laminates exhibit any voids. Unbonded panel/laminate repairs are discussed in a separate section of this report.

Twenty-four autoclave loads of boron-epoxy laminates and the associated process control specimens were cured during Phase III. The laminate quality was verified by normal process control specimens cut from flat laminate panels. The bond of interleaved titanium shims within the laminates was verified by standard titanium lap shear specimens. In addition to these, double-lap cocured boron-epoxy to titanium specimens were made with each autoclave load and tested for further verification.

The boron-to-titanium lap shear specimens, illustrated in Figure 56, were prepared by cocuring 4 plies of boron-epoxy prepreg to each side of two butt joined titanium plates. The individual specimens were cut from the resulting assembly following its cure. One layer of supported EA9601 adhesive film was applied to each bondline and cocured with

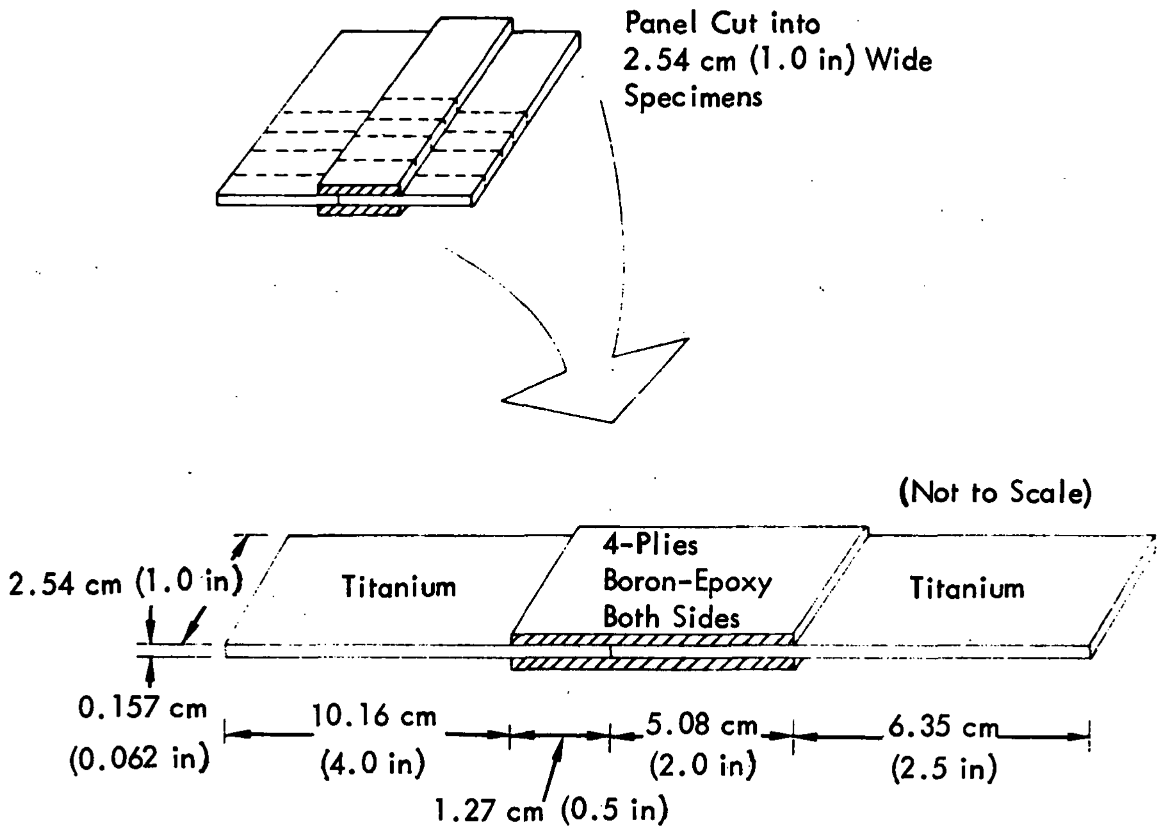


FIGURE 56.- COCURED BORON-EPOXY-TO-TITANIUM LAP SHEAR TEST SPECIMEN

the boron-epoxy prepreg. Each boron-epoxy lay-up was covered with one ply of porous teflon coated fiberglass, followed by one ply of 120 fiberglass cloth. Each titanium plate was supported by an 0.508 mm (0.020 in.) thick shim during cure to ensure proper alignment after cure. In addition, the boron-epoxy prepreg lay-ups were laterally supported to prevent fiber washout during cure. This assembly was then cured along with the represented load of boron-epoxy laminates. Individual lap shear specimens were machined from the cured assembly and tested for lap shear strength. The resulting data are representative of the boron-epoxy to titanium shim bond and supplement the standard titanium finger panel process control lap shear data.

Because of the bulk of the data accumulated from process control specimens associated with the boron-epoxy laminates, the titanium shim-to-laminate bond, and the laminate-to-stringers-and-panels bond, each table is presented in two separate parts. The first part presents the data in SI units and the second part presents the data in U.S. customary units.

Table IV shows the summary of the process control data for verification of laminate quality in each of the 24 autoclave runs. The process control data for verification of the boron-epoxy-to-titanium shim bonds in each of the 24 autoclave runs is shown in Table V.

A total of 39 bond cycles were conducted in completing the bonding of laminates to stringers and panels. All major difficulties associated with cool tool usage were overcome, but, even so, the process remains a relatively tedious, painstaking one which could be greatly simplified by the use of matched tools. Such tooling would be easily justified in a normal production run but would have proven too costly for building only three boxes. Process verification test data for all bond cycles are shown in Table VI for both the standard finger panels and for the boron laminate-to-aluminum specimen. The precured boron-epoxy-to-aluminum shear test specimen is shown in Figure 57.

5.3 MATERIAL REVIEW BOARD

Throughout the fabrication process, when a part or process deviated from strict compliance to specification or drawing requirements, such deviation was documented on a Discrepancy Report (DR). The non-conformance was then reviewed by one or more highly qualified specialists from the appropriate technical discipline for disposition. These specialists comprise the Material Review Board (MRB). DR's are reviewed by the AFPRO Quality Assurance Branch and signed if concurrence is granted to the MRB disposition.

Dispositions of discrepant parts by the MRB fall into three general categories:

- Use as is - Minor deviations in areas where good structural margins-of-safety exist or in areas where the impact is insignificant.
- Repair & use - Repairable defects where equivalent strength/endurance can be replaced by repair.

TABLE IV.- PART 1 - SUMMARY OF BORON-EPOXY LAMINATE PROCESS CONTROL DATA

Test		1		2		3		4		5		6		7		8		9		10		11	
	Autoclave Load No.	1		2		3		4		5		6		7		8		9		10		11	
	Laminate Width (cm)	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29
	Tape Lot No.	59	59	59	59	59	59	59	59	59	59	59	59	59	-	59	59	59	59	59	59	59	59
	Tape Roll No.	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	89	169	89	120
Longitudinal (0°) Flexure △1	High	1.72	1.66	2.09	2.01	2.06	1.98	2.02	2.07	2.02	2.03	1.94	2.01	1.98	No 2.24 cm Laminates This Run	1.89	1.83	1.92	2.01	1.99	1.93	1.95	1.98
	Avg	1.66	1.60	1.96	1.85	2.03	1.89	1.98	2.05	1.98	1.97	1.81	1.99	1.94		1.87	1.80	1.84	1.87	1.95	1.91	1.91	1.91
	Low	1.55	1.56	1.77	1.66	2.02	1.85	1.93	2.02	1.96	1.92	1.67	1.97	1.90		1.83	1.78	1.79	1.68	1.90	1.88	1.85	1.85
Horizontal (0°) Shear △2	High	84.1	79.9	107	95.2	103	100	106	108	106	106	99.3	106	102		102	95.8	115	98.6	112	104	102	92.4
	Avg	80.6	79.3	103	93.8	102	98.6	105	105	103	105	97.9	104	96.5		101	95.1	112	97.2	110	100	97.9	91.0
	Low	77.9	79.3	101	91.7	101	97.9	104	103	101	105	95.8	96.5	86.2		100	94.5	109	95.8	109	93	91.0	90.3
		△3	△3																				

Load No.		12			13			14		15		16		17		18		19		20		21		22		23		24	
Width (cm)		5.08	5.08	2.29	5.08	2.29	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29	5.08	2.29
Lot No.		59	60	59	60	59	60	60	60	60	60	60	60	60	60	60	61	60	61	60	60	-	60	60	61	60	61	60	61
Roll No.		86	137	175	61	175	110	154	37	51	117	66	127	157	126	252	66	264	45	284	97	-	168	84	67	270	48	254	111
0° Flex	High	1.95	2.02	2.04	1.98	2.07	2.01	2.13	2.07	1.99	2.04	1.68	1.95	2.04	1.95	No 5.08 cm Laminates This Run	2.06	1.88	1.76	1.84	1.75	2.04	2.10	1.98	2.07	1.91	1.98	1.81	1.76
	Avg	1.89	1.82	1.93	1.91	2.01	1.98	2.04	2.00	1.91	1.92	1.56	1.89	1.98	1.94		2.02	1.82	1.75	1:69	1.66	1.99	2.05	1.84	1.95	1.84	1.96	1.80	1.73
	Low	1.81	1.63	1.87	1.82	1.92	1.93	1.98	1.96	1.86	1.91	1.49	1.80	1.93	1.92		1.99	1.74	1.73	1.53	1.61	1.97	2.01	1.66	1.85	1.77	1.95	1.79	1.70
0° Shear	High	105	107	100	112	108	105	108	109	108	104	86.9	105	108	108		113	94.5	101	92.4	88.3	109	108	101.4	97.9	106	93.8	103	96.5
	Avg	104	101	99.3	107	101	103	103	105	105	101	85.5	100	88.9	104		112	93.1	101	91.0	86.2	104	105	98.6	94.5	104	93.1	102	93.1
	Low	102	93.1	97.2	105	89.6	101	99.2	98.6	101	99.3	83.4	97.2	75.8	101		111	90.3	97.9	69.6	84.8	95.8	101	94.5	92.4	103	91.0	101	89.6
												△3		△3						△3	△3								

Notes: △1 Minimum specification requirements for 0° Flexure of boron-epoxy laminates are: Average - 1.65 GN/m^2; Individual - 1.55 GN/m^2

△2 Minimum specification requirements for 0° Horizontal Shear of boron-epoxy laminates are: Average - 89.6 MN/m^2; Individual - 75.8 MN/m^2

△3 The few low values exhibited in the autoclave runs were accepted in MRB actions: those in run 1 were accepted based on results of destructive tests, those in runs 16, 17, and 20 were accepted based on results of the accompanying B-to-Ti cocured specimens.

TABLE IV.- PART 2 - SUMMARY OF BORON-EPOXY LAMINATE PROCESS CONTROL DATA

Test	Autoclave Load No.	1		2		3		4		5		6		7		8		9		10		11	
	Laminate Width (In.)	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9
	Tape Lot No.	59	59	59	59	59	59	59	59	59	59	59	59	59	-	59	59	59	59	59	59	59	59
	Tape Roll No.	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	89	169	89	120
Longitudinal (0°) Flexure (1)	High	248.8	241.0	303.3	291.5	298.6	286.9	293.3	300.8	292.6	293.8	281.1	291.8	287.1	No 0.9" Laminates This Run	274.1	266.0	278.1	291.4	289.2	280.0	283.1	286.9
	Avg	240.3	231.6	283.6	268.5	294.8	274.7	287.0	297.6	286.7	285.7	262.5	288.4	281.5		270.5	261.4	267.6	271.4	282.1	276.6	277.0	277.4
	Low	224.2	226.1	256.0	241.1	292.7	269.0	280.0	293.4	283.8	277.9	243.0	285.3	275.8		265.1	257.9	260.3	240.0	275.1	272.9	268.5	268.7
Horizontal (0°) Shear (2)	High	12.2	11.6	15.5	13.8	15.0	14.5	15.4	15.6	15.4	15.4	14.4	15.4	14.8		14.8	13.9	16.7	14.3	16.3	15.1	14.8	13.4
	Avg	11.7	11.5	15.0	13.6	14.8	14.3	15.2	15.2	15.0	15.3	14.2	15.1	14.0		14.6	13.8	16.3	14.1	16.0	14.5	14.2	13.2
	Low	11.3	11.5	14.6	13.3	14.6	14.2	15.1	14.9	14.7	15.2	13.9	14.7	12.5		14.5	13.7	15.8	.13.9	15.8	13.5	13.2	13.1
		(3)	(3)																				

Load No.		12			13			14		15		16		17		18		19		20		21		22		23		24	
Width (In.)		2.0	2.0	0.9	2.0	0.9	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9	2.0	0.9
Lot No.		59	60	59	60	59	60	60	60	60	60	60	60	60	60	60	61	60	61	60	60	-	60	60	61	60	61	60	61
Roll No.		86	137	175	61	175	110	154	37	51	117	66	127	157	126	252	66	264	45	284	97	-	168	84	67	270	48	254	111
0° Flex	High	282.7	292.9	295.5	287.1	300.9	292.0	308.4	300.1	288.8	296.1	243.9	283.3	295.3	283.1	No 2.0" Laminates This Run	298.1	272.0	255.8	266.3	253.4	294.5	304.6	287.2	300.4	276.8	287.7	262.8	254.6
	Avg	273.8	268.8	280.5	276.3	290.9	286.7	295.5	290.7	277.7	278.8	225.8	274.0	287.1	281.5		293.1	263.7	253.7	244.8	241.2	289.1	296.7	266.9	283.5	267.3	284.3	261.5	250.8
	Low	262.9	236.9	271.5	264.4	277.8	279.4	287.0	284.9	269.2	277.2	216.6	261.1	280.1	278.8		289.0	252.0	250.7	221.2	232.9	285.7	290.9	240.5	268.1	256.9	282.2	260.3	247.0
0° Shear	High	15.3	15.6	14.5	16.2	15.6	15.3	15.7	15.8	15.6	15.1	12.6	15.2	15.6	15.7		16.4	13.7	14.6	13.4	12.8	15.8	15.6	14.7	14.2	15.4	13.6	15.0	14.0
	Avg	15.1	14.7	14.4	15.5	14.6	15.0	15.0	15.2	15.3	14.7	12.4	14.5	12.9	15.1		16.2	13.5	14.7	13.2	12.5	15.1	15.3	14.3	13.7	15.1	13.5	14.8	13.4
	Low	14.8	13.5	14.1	15.2	13.0	14.6	14.4	14.3	14.7	14.4	12.1	14.1	11.0	14.6		16.1	13.1	14.2	13.0	12.3	13.9	14.7	13.7	13.4	14.9	13.2	14.6	13.0
												(3)		(3)						(3)	(3)								

Notes: (1) Minimum specification requirements for 0° Flexure of boron-epoxy laminates are: Average - 240 KSI; Individual - 225 KSI

(2) Minimum specification requirements for 0° Horizontal Shear of boron-epoxy laminates are: Average - 13 KSI; Individual - 11 KSI

(3) The few low values exhibited in the autoclave runs were accepted in MRB actions: those in run 1 were accepted based on results of destructive tests, those in runs 16, 17, and 20 were accepted based on results of the accompanying B-to-Ti cocured specimens.

TABLE V. - PROCESS CONTROL DATA FOR BOND OF BORON-EPOXY TO TITANIUM SHIMS

PART I

Specimen Type	Autoclave Load	Shear Strength (MN/m^2) 1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
Ti-to-Ti △1	Low	23.2	28.4	22.0	19.5	17.7	29.1	20.9	37.6	40.5	42.3	38.2	39.9	44.2	46.4	41.6	Passed Minimum	39.5	40.0	Specimen Not Made	36.0	45.7	37.5	38.3	40.8
Std. Finger	Avg	25.2	29.6	24.1	20.8	20.8	30.1	21.6	39.8	42.9	43.9	40.3	40.7	46.4	47.2	43.9		39.7	40.4		39.8	46.8	39.7	42.1	41.4
Panels △2	High	27.0	31.2	27.9	22.0	22.3	30.8	22.4	41.7	43.4	45.3	42.6	42.1	46.3	47.8	45.7		35.1	40.7		41.4	48.2	40.4	44.0	45.8
B-to-Ti	Low	Specimen Not							42.1	43.4	48.8	37.9	39.9	37.9	39.7	45.6	31.9	44.7	Adhesive Omitted	35.2	41.1	42.3	43.8	Specimens Incorrect	41.5
Cocured	Avg	Used Prior							47.9	47.2	51.3	40.7	45.3	41.8	45.9	47.1	43.3	48.0		36.8	44.1	46.8	44.0		45.2
Specimen △3	High	to Run 8							50.8	49.2	53.6	44.6	47.6	45.6	49.3	40.7	47.2	50.2		36.9	46.8	50.0	41.4		48.7

PART II

Specimen Type	Autoclave Load	Shear Strength (psi) 1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
Ti-to-Ti △1	Low	3360	4120	3190	2800	2560	4227	3030	5450	5880	6140	5540	5793	6412	6725	6034	Passed Minimum	5682	5800	Specimen Not Made	5190	6571	5409	5555	5910
Std. Finger	Avg	3660	4300	3490	3010	3010	4371	3130	5770	6220	6360	5850	5906	6676	6850	6367		5760	5866		5720	6788	5714	6057	6300
Panels △2	High	3910	4530	4010	3190	3320	4461	3250	6050	6290	6570	6180	6063	6714	6936	6627		5809	5900		6000	6984	5809	6380	6590
B-to-Ti	Low	Specimen Not							6110	6290	7080	5500	5789	5500	5762	6609	4630	6490	Adhesive Omitted	5100	5960	6080	6307	Specimens Incorrect	6020
Cocured	Avg	Used Prior							6940	6840	7440	5900	6519	6058	6659	6824	6285	6968		5340	6400	6788	6382		6550
Specimen △3	High	to Run 8							7370	7140	7770	6470	6907	6564	7148	7058	6850	7285		5631	6790	7190	6500		7060

Notes: △1 In Runs 1-7, a combination of a high-flow adhesive and specimen size/configuration caused erratic test results. Tests showed this condition to be peculiar to the process control specimens. The laminates were satisfactory. A different adhesive was used for Runs 8-24.

△2 The minimum desired shear strength using this specimen configuration is 28.3 MN/m^2 (4100 psi). The lower values of runs 1 through 7 are not representative of actual bond strength in the laminates.

△3 No requirement established.

TABLE VI. - PART 1 - SHEAR TEST RESULTS FOR COOL TOOL BONDING LOADS

Specimen Type	Cool Tool Load	Shear Strength (MN/m^2) 1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
Al-to-Al Std Finger Panels (3)	Low	34.5	Verifilm Run - No Data	33.9	34.2	Specimens Cured Without Pressure	41.0	Scrapped Panel	27.0	39.1	33.6	41.5	42.4	40.0	38.4	35.5	44.8	40.5	41.0	35.7	36.4	37.9	38.5	37.8	39.7
	Avg	36.6		35.0	35.1		42.3		32.8	40.4	35.8	42.8	44.2	41.1	39.8	46.3	46.3	41.2	43.0	40.1	38.2	41.1	39.8	40.4	41.3
	High	38.2		36.1	37.1		43.7		37.3	41.5	37.2	45.1	45.8	42.7	41.1	36.7	47.4	42.2	43.9	43.0	40.8	42.7	41.1	42.2	42.5
Precured Boron-to-Al Double Lap (4) Shear Specimens	Low	None Made		30.6	34.0		24.8		29.6	33.5	27.0	33.8	30.7	20.9	33.0	32.9	31.5	33.2	31.5	42.1	32.4	34.9	7.10	34.0	33.2
	Avg			31.1	34.4		26.6		30.9	34.4	31.9	35.2	33.4	21.9	33.4	33.2	33.3	34.1	32.8	42.8	33.2	35.7	7.93	35.1	34.8
	High			32.0	34.7		28.3 (1)		33.9	35.4	34.2	36.0	35.3	22.6 (1)	35.6	33.6	34.3	35.0	34.1	43.6	34.0	36.4	8.55 (1)	36.5	37.1

Specimen Type	Cool Tool Load	Shear Strength (MN/m^2) 25	26	27	28	29	30	31	32	33	34	35	36	37	38	39
Al-to-Al Std Finger Panels	Low	39.0	39.5	39.6	40.8	39.7	41.4	39.4	34.4	34.2	38.3	36.1	33.1	35.4	32.3	37.4
	Avg	41.3	40.6	40.2	41.9	40.4	42.0	40.0	35.8	38.8	39.9	38.7	35.2	38.6	35.4	39.2
	High	42.7	41.1	41.1	42.4	41.2	42.7	40.6	36.8	42.1	41.1	40.5	37.2	41.0	37.9	41.9
Precured Boron-to-Al Double Lap Shear Specimens	Low	32.5	34.5			36.0								34.8	34.8	35.5
	Avg	34.1	35.1	(2)	(2)	37.7	(2) ←	→	→	→	→	→	(2)	35.5	35.2	35.1
	High	35.7	35.5			38.0								36.0	35.9	35.8

Notes: (1) Improper test specimen preparation.

(2) The precured boron laminate for these specimens was contaminated with oil during machining. Despite repeated attempts to clean, the test specimens were not satisfactory and data are not valid.

(3) The minimum desired shear strength using this specimen configuration is 24.1 MN/m^2.

(4) No requirement established.

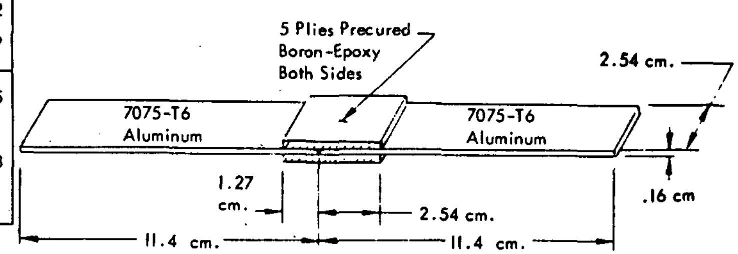


FIGURE 57. - PART 1 - PRE-CURED BORON-EPOXY TO ALUMINUM LAP SHEAR TEST SPECIMEN

TABLE VI. - PART 2 - SHEAR TEST RESULTS FOR COOL TOOL BONDING LOADS

Specimen Type	Cool Tool Load	Shear Strength (psi) 1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
Al-to-Al	Low	5000	VERIFILM RUN - NO DATA	4920	4960	Specimens Cured Without Pressure	5940	Scrapped Panel	3920	5670	4880	6020	6160	5800	5576	5143	6500	5877	5940	5180	5280	5500	5580	5480	5760
Std Finger	Avg	5309		5030	5050		6130		4750	5860	5200	6210	6406	5966	5772	5199	6710	5972	6190	5810	5536	5957	5772	5866	5987
Panels △3	High	5547		5200	5380		6340		5410	6020	5400	6540	6640	6200	5961	5320	6880	6120	6370	6240	5920	6200	5960	6120	6160
Precured	Low	NONE MADE		4410	4930		3590		4290	4860	3914	4903	4454	3035	4783	4768	4567	4811	4570	6100	4700	5020	1030	4935	4809
Boron-to-Al	Avg			4510	4990		3860		4480	4987	4624	5103	4850	3175	4846	4818	4834	4941	4759	6206	4816	5171	1150	5093	5040
Double Lap △4	High			4610	5030		4100		4910	5137	4954	5221	5116	3278	5158	4871	4969	5083	4948	6320	4930	5281	1240	5297	5385
Shear Specimens							△1							△1									△1		

Specimen Type	Cool Tool Load	Shear Strength (psi) 25	26	27	28	29	30	31	32	33	34	35	36	37	38	39
Al-to-Al	Low	5660	5731	5740	5920	5760	6000	5714	4990	4960	5560	5240	4800	5130	4680	5430
Std Finger	Avg	5990	5807	5836	6076	5853	6093	5799	5185	5630	5790	5620	5070	5600	5138	5688
Panels	High	6200	5961	5960	6145	5980	6200	5885	5330	6100	5960	5870	5400	5940	5500	6080
Precured	Low	4708	5000			5228								5050	5050	5150
Boron-to-Al	Avg	4941	5092	△2	△2	5419	△2 ←						→ △2	5151	5110	5090
Double Lap	High	5178	5143			5511								5175	5210	5190
Shear Specimens																

Notes:
△1 Improper test specimen preparation.
△2 The precured boron laminate for these specimens was contaminated with oil during machining. Despite repeated attempts to clean, the test specimens were not satisfactory and data are not valid.
△3 The minimum desired shear strength using this specimen configuration in 3500 psi.
△4 No requirement established.

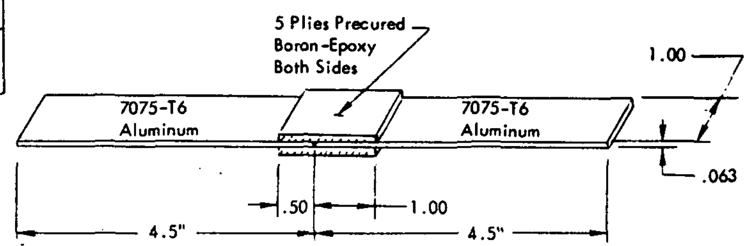


FIGURE 57. - PART 2 - PRE-CURED BORON-EPOXY TO ALUMINUM LAP SHEAR TEST SPECIMEN

- Scrap – Non-repairable defects, or defects that are too costly to repair.

During the fabrication of the three center wing boxes, a total of 174 deviations were documented. Of these, 31 were due to laminate or laminate hole quality, 40 were due to bonding related deviations, and 92 were attributed to metal parts and processing/assembly. Eleven were attributed to test specimens/tooling/materials. A more complete listing of MRB actions during the production program is included in Appendix C.

6.0 SOLUTIONS TO MANUFACTURING PROBLEMS

During production of the composite-reinforced wing boxes, several difficulties were encountered which had not been identified during the advanced development program phase. These were primarily related to the use of "shop-aid" or prototype tooling (rather than the more sophisticated tools which would be used in a full production run) and to unanticipated variations in production materials. The resolution of the more important of these is discussed below.

6.1 ADHESIVE BOND OF INTERLEAVED TITANIUM SHIMS

Process control specimens, made with the first seven sets of laminates, indicated sizeable and erratic variations in laminate-to-titanium shim adhesive strength. Data from two of the autoclave loads showed lap shear strengths better than specification requirements but lower than normally attained. The other five sets of P.C. data did not meet specification minimums. The fabrication of boron-epoxy laminate assemblies was suspended due to this failure of process control titanium finger panels to meet specification lap shear requirements, and a thorough investigation was conducted to determine the cause of failure.

After extensive study, this difficulty was traced to a combination of a high-flow adhesive and the process control specimen size/configuration, and was concluded to be peculiar to the process control test specimen. Subsequent tests of full laminate specimens verified this conclusion and laminate production was resumed, with an equally qualified adhesive which had a lesser flow. The investigations followed a dual path, as discussed below.

6.1.1 Process Control Specimen/Adhesive Investigations

The failure of titanium process control specimens (finger-panels) to meet lap shear specification requirements required a careful assessment of adhesive used, primer used, specimen metal cleaning, specimen design, and bond thickness. These investigations covered all aspects of primer/adhesive application and processing. Titanium adherends (6Al-4V) with overlaps of 1.63 cm (0.64 in.) were processed, cured at two combinations of temperature and pressure, and tested at room temperature to provide data for the investigation. The detailed investigation covered processing, material, and simulated component tests. Comparative lap shear specimens were prepared, tested, and evaluated as follows:

- o Primer application by personnel in two separate areas of the plant: no significant difference.
- o Three primer drying cycles: no significant difference.
- o Two primer thicknesses: no significant difference.
- o Titanium cleaning by the production facility versus manufacturing research facility: no significant difference.

- Flow reduction:

  By Aging prior to adhesive cure

  Lap shear increases with increased open time up to 10 days

  By Specimen Configuration (Blister Panel - See Table VII)

  Increases lap shear in comparison with finger panel

- Bondline thickness: Lap shear increases with increase in bondline thickness within the range tested.

- Adhesive film weight: Film weight of 0.045 psf had lap shear strength below specification; film weight of 0.060 had lap shear strength above specification. This result is in keeping with the results obtained from bondline thickness above.

- Different primers: FM123 adhesive showed no difference in lap shear strength when primed with BR123 or BR125 primers; however, a considerable drop in shear strength resulted from priming with EC3921.

- Other adhesives qualified to STM30-102: AF127-3 adhesive showed lap shear strength considerably below the specification minimum. Both FM1000 and EA9601 adhesives had lap shear strengths considerably above specification with EA9601 values the highest.

- Two cure cycles: In general, when the FM123 adhesive was cured at 394°K (250°F) under 0.207 MN/m^2 (30 psi) pressure, lap shear strengths near the specification value were obtained. When cured in accordance with the boron cure cycle, 450°K (350°F) under 0.586 MN/m^2 (85 psi), lap shear strengths were below specification.

Comparative lap shear strengths for the adhesives tested, including two film weights for FM123 adhesive, are shown in Figure 58. The analysis of the lap shear test data indicated that the resulting shear stress properties were directly related to the specimen bondline thickness; this property being dictated by the flow of the adhesive. Additional data gained from specimens cut from components fabricated during the process represented by low process control lap shear properties showed that acceptable bonds had, in fact, been achieved in the component, despite the low properties indicated by the process control coupons. Since the adhesive performance was thus isolated to the process control coupons, corrective action was directed to this area.

This difficulty was confined to the finger panels used for process control as illustrated in Table VII. In the cocured boron-epoxy to titanium specimens, which represent the actual laminate condition, both of the adhesives tested performed satisfactorily. Blister detection panels, which tend to limit the adhesive flow, also indicated acceptability of both adhesive systems. The finger panel data, however, show a shear strength with the EA 9601 which is more representative of values achieved in the actual bond within the laminate. Since either adhesive was qualified under existing specifications, it was judged expedient to continue to use the finger panels for process verification with a change to EA 9601 as the shim-bonding adhesive.

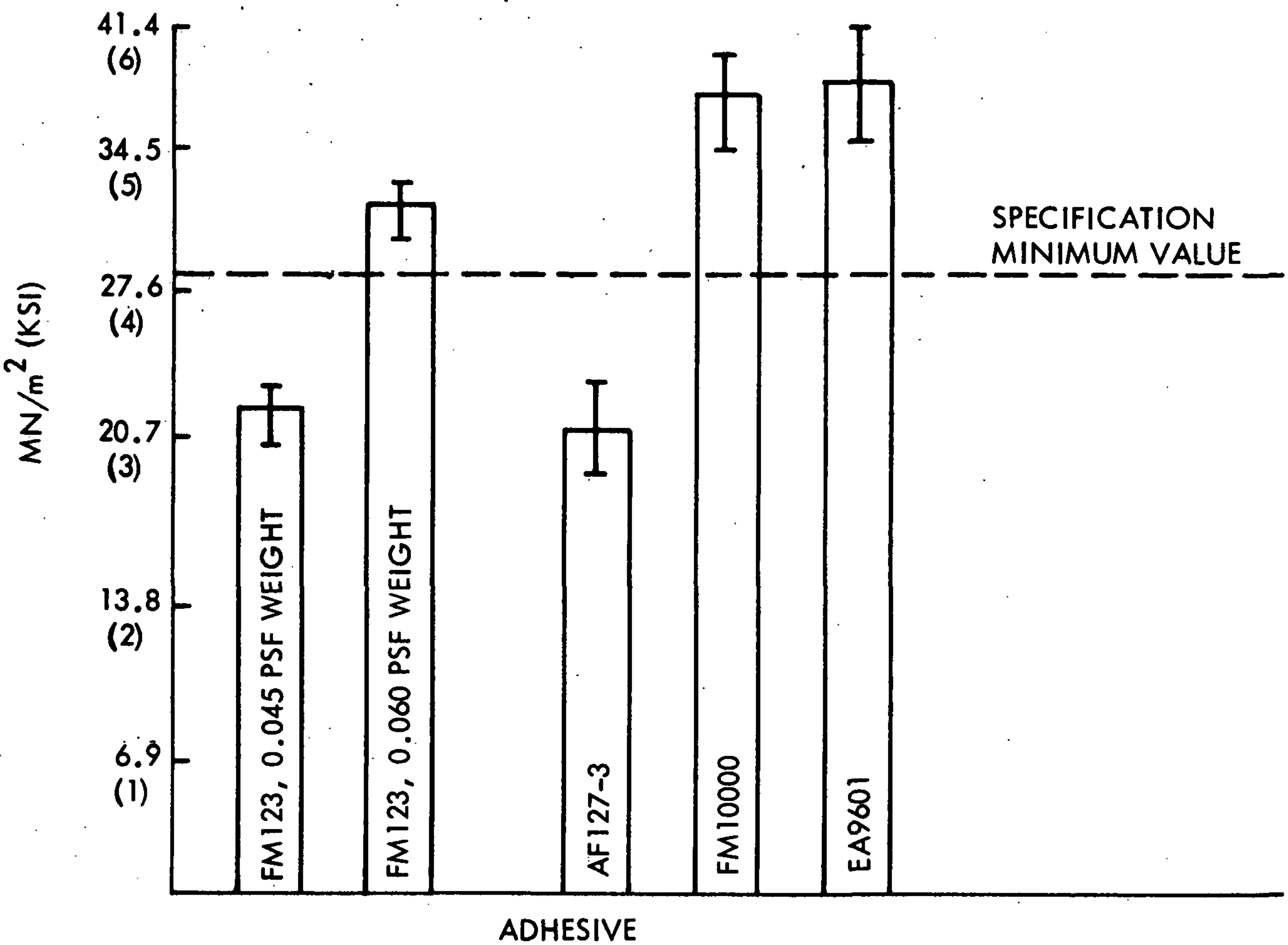


FIGURE 58 - LAP SHEAR DATA FOR ADHESIVES

For subsequent autoclave laminate runs, in addition to the finger panels, cocured boron-epoxy/titanium specimens were fabricated with each autoclave load of boron-epoxy laminates. These specimens were cut and tested to provide supporting process control data.

6.1.2 Acceptability of Existing Laminates

As additional verification that the low lap shear results were only a test panel problem that did not exist within the actual laminate, and that the laminates already made were acceptable for use, three of the laminates from two autoclave runs were sacrificed in tests. Four test articles, designated as JE-5, JE-6, JE-7-1, and JE-7-2, were fabricated and tested to verify the load transfer capability of the shims/laminate at the ends of the laminate. The tests exceeded the airplane load requirements by a sizeable margin, demonstrating that the remaining laminates were acceptable for use.

TABLE VII.- ADHESIVE PROPERTY/SPECIMEN DATA

Specimen Type \ Adhesive / Primer		EA 9601 EA 9201.1		FM 123-4 BR 123		Remarks
		MN/m^2	psi	MN/m^2	psi	All specimens cured in standard autoclave cure cycle for laminates
Finger Panel	Low =	35.79	5190	24.27	3520	1.6 cm (5/8") overlap; Ti-to-Ti 0.102 cm (0.040")
	Avg =	37.92	5500	25.65	3720	
	High=	38.75	5620	26.75	3880	
Boron to Titanium	Low =	35.08	5088	34.86	5056	1.6 cm (5/8") boron double lap on Ti adherends - Cocured - Metal Failure
	Avg =	35.19	5104	34.97	5072	
	High=	35.38	5128	35.30	5120	
Blister Detection Panels	Low =	33.79	4900	30.81	4468	This standard blister panel consists of two sheets bonded together from which test specimens and desired overlaps are then machined.
	Avg =	35.03	5080	31.60	4583	
	High=	35.85	5200	32.32	4687	

Test article laminates were bonded to metal plates with cool-tool techniques and prepared for testing by secondary bonding of doublers and loading tabs, and application of strain gages. The JE-5 test component represented the skin laminate run-out configuration for stringer #18 on the lower surface at W.S. 200-220. This component was successfully tested to a tension load which exceeded airplane ultimate load requirements by approximately 30%. Failure occurred by net section fracture of the aluminum skin panel.

The JE-6 test component illustrated in Figures 59 and 60 simulated the skin laminate run-out configuration for stringer #18 on the lower surface at the W.S. 140 cut-out. This component was successfully tested to a tension load which exceeded airplane ultimate load requirements by approximately 42%. Failure occurred by net section fracture of the boron-epoxy laminate at a fourth fastener location near the laminate end. Strain data showed (Figure 61) that measured strains in the proximity of the failure were equal to, or exceeded, the boron-epoxy design ultimate tensile strength of 1.24 GN/m^2 (180 KSI).

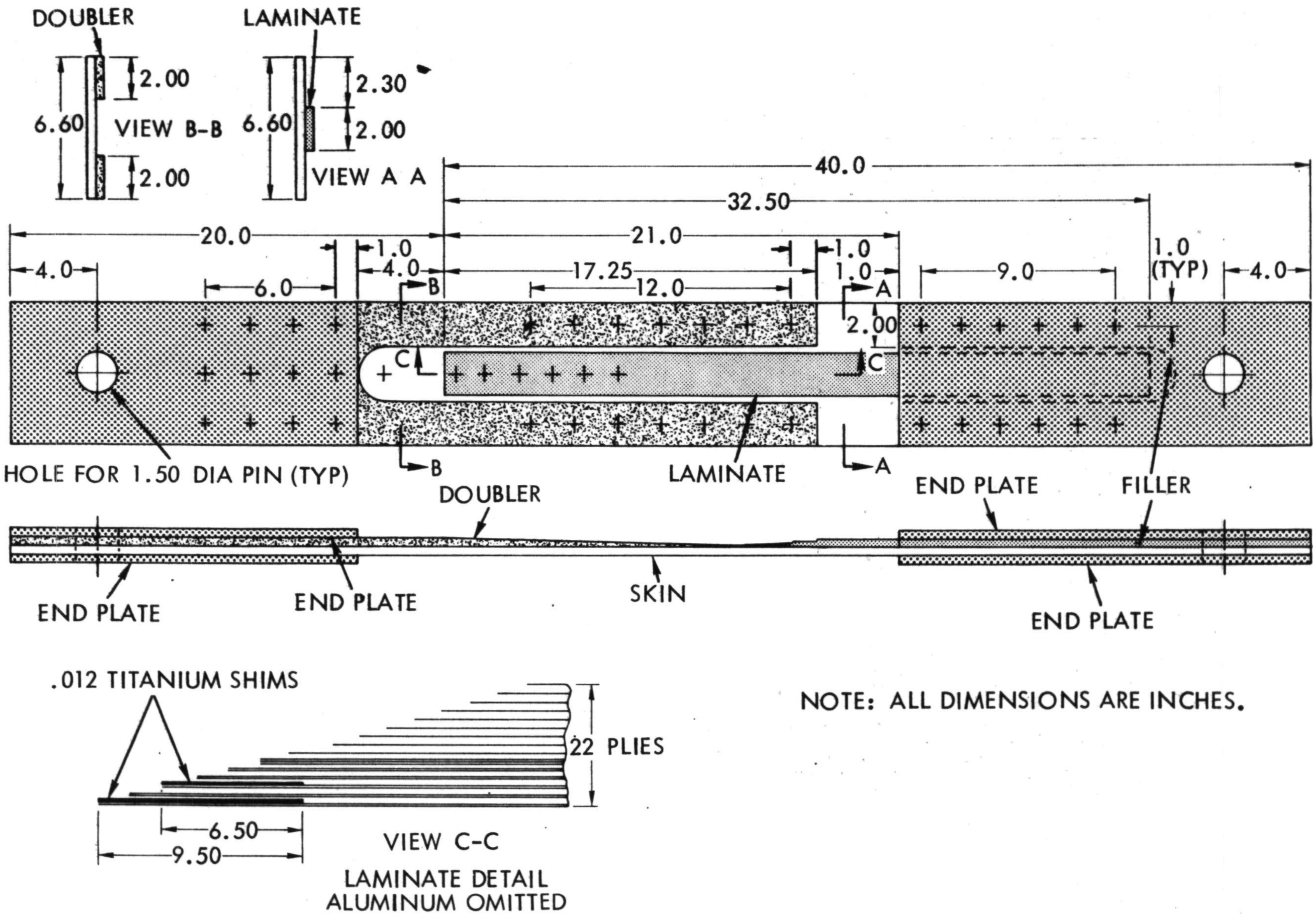
The JE-7 specimens were identical in configuration to JE-5 but used a laminate from a different autoclave run. Test results and failure modes were almost identical to those obtained from the JE-5 specimen. Table VIII summarizes the test results.

TABLE VIII. - JE-5, -6, & -7 TEST EVALUATION

Item	Component							
	JE-5		JE-6		JE-7-1		JE-7-2	
	MN/m^2	ksi	MN/m^2	ksi	MN/m^2	ksi	MN/m^2	ksi
Calculated Stress at ⚠1 Failure Load - Aluminum	361.2	52.4	410.2	59.5	361.2	52.4	351.6	51.0
Boron	1061.1	153.9	1257.6	182.4	1061.1	153.9	1034.3	150.0
Required Airplane ⚠2 Ultimate Stress - Aluminum	275.8	40.0	289.6	42.0	275.8	40.0	275.8	40.0
Boron	799.8	116.0	896.4	130.0	799.8	116.0	799.8	116.0

⚠1 Stresses calculated at point of failure.

⚠2 Includes maximum thermal residual stress.



NOTE: ALL DIMENSIONS ARE INCHES.

FIGURE 59. - 130-JE-6 DESIGN

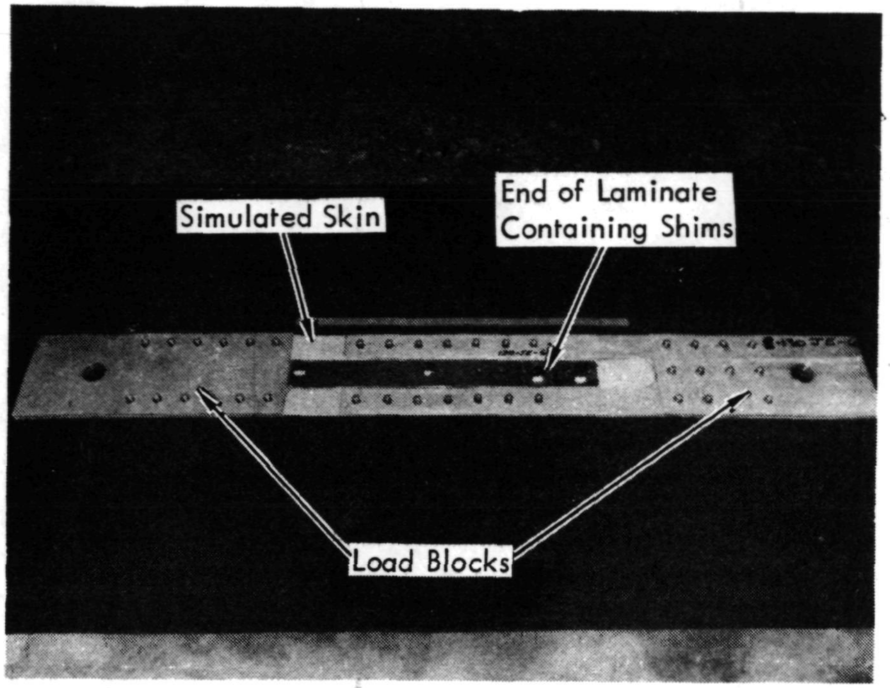


FIGURE 60. - 130-JE-6 TEST SPECIMEN

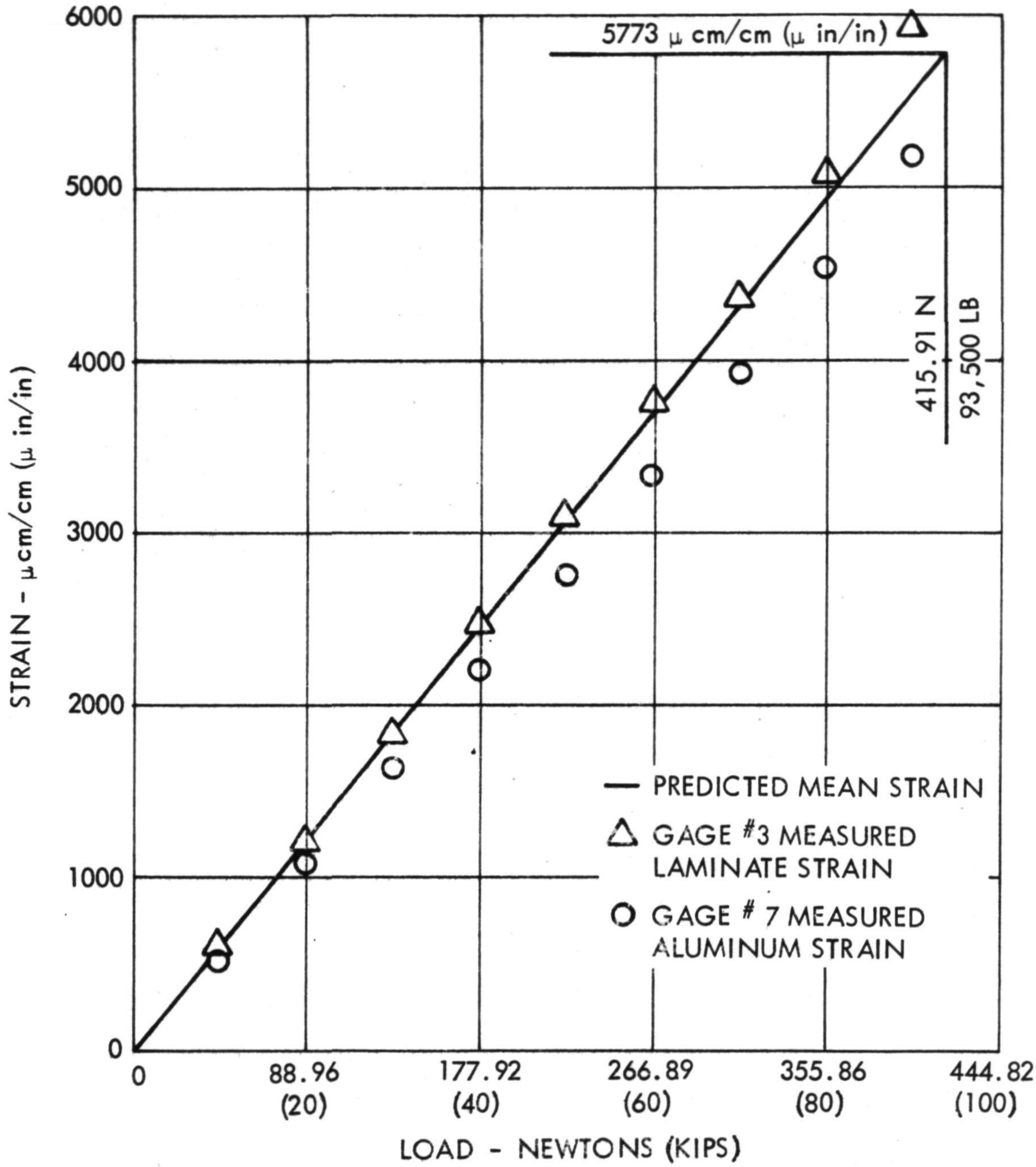


FIGURE 61. - 130-JE-6 STRAIN DATA

Prior to testing, each specimen was instrumented with eight electrical resistance type strain gages, and each specimen was fitted with a lateral support fixture. Eight axial strain gages were applied to each specimen. The support fixture was clamped to the specimens as illustrated in Figures 62 and 63. Teflon was used between the specimen and spanwise support bars to minimize friction. Since the specimens were slightly bowed, straightening by the support arrangement prior to test provided a better representation of a skin-stringer combination.

All tests were conducted in a universal testing machine, using standard fork sets to allow pin loading. A photograph of the test arrangement is shown in Figure 64.

Typical test specimen failures are illustrated in Figures 65 through 68.

6.2 PROVIDING QUALITY LAMINATE HOLES

The pre-punched holes in the laminates were "plugged" during laminate cure to maintain shim alignment and to prevent their being filled with cured resin. Short pieces of wooden dowel pins were first used, but these were difficult and time consuming to cut to the proper length. These were soon replaced by short lengths of teflon dowels; however, it was discovered that they allowed the boron plies and titanium doublers to shift during the curing cycle resulting in shim misalignment in the holes.

This misalignment was not a severe problem but it acted as a catalyst to create a number of poor holes. The reamer being used to clean up the fastener holes was very sensitive to lateral forces caused by striking a shim. The lateral movement of the reamer, and the smearing of the shim, combined to produce a large variation in hole quality. Some holes were good, some repairable, and some quite poor.

The shims were immobilized, and the difficulty resolved, by replacing one of the teflon pins with a steel pin, providing a significant improvement in hole quality.

6.3 COOL-TOOL IMPROVEMENT

Thermal expansion of the first full bonding load of five stringers caused the end restraint blocks on the cool tool to deflect, allowing the ends of the stringers to move upward and crippling the stringer ends. Some were usable because the crippled ends were in an excess area which was later machined away, but one or two had to be scrapped. In order to eliminate this deflection, several dowel pins and bolts were added to the restraint block and between the longitudinal beams and the top surface plate of the cool tool. No further problems with end block deflection occurred.

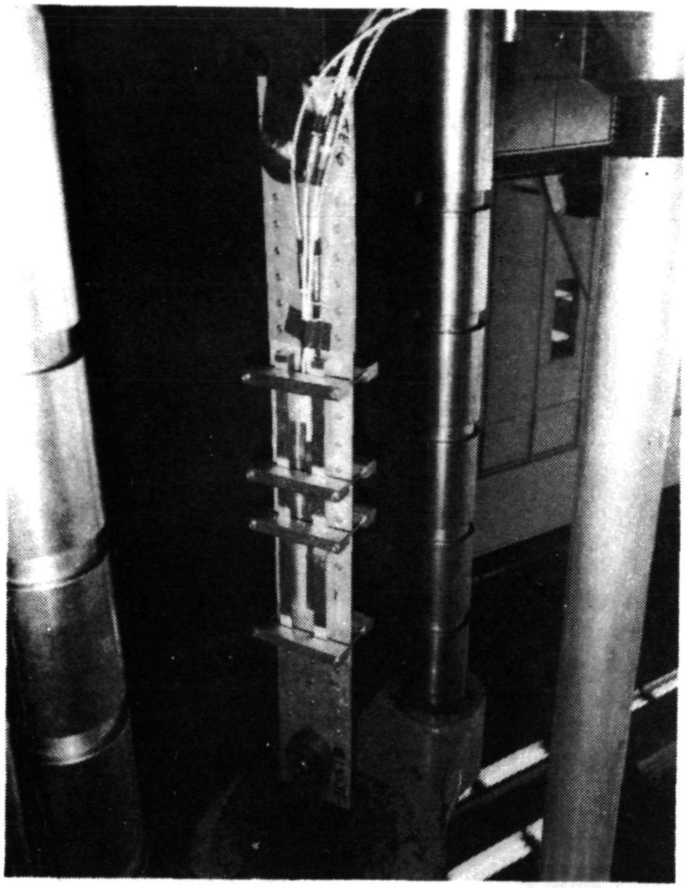


FIGURE 62. - SUPPORT FIXTURE ARRANGEMENT ON SKIN SIDE OF SPECIMEN

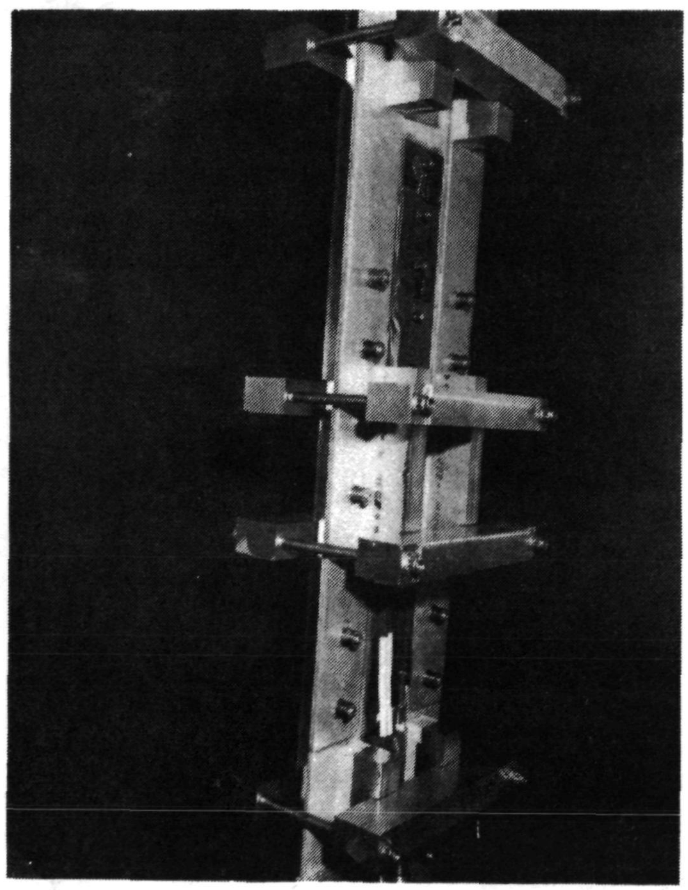


FIGURE 63. - SUPPORT FIXTURE ARRANGEMENT ON LAMINATE SIDE OF SPECIMEN

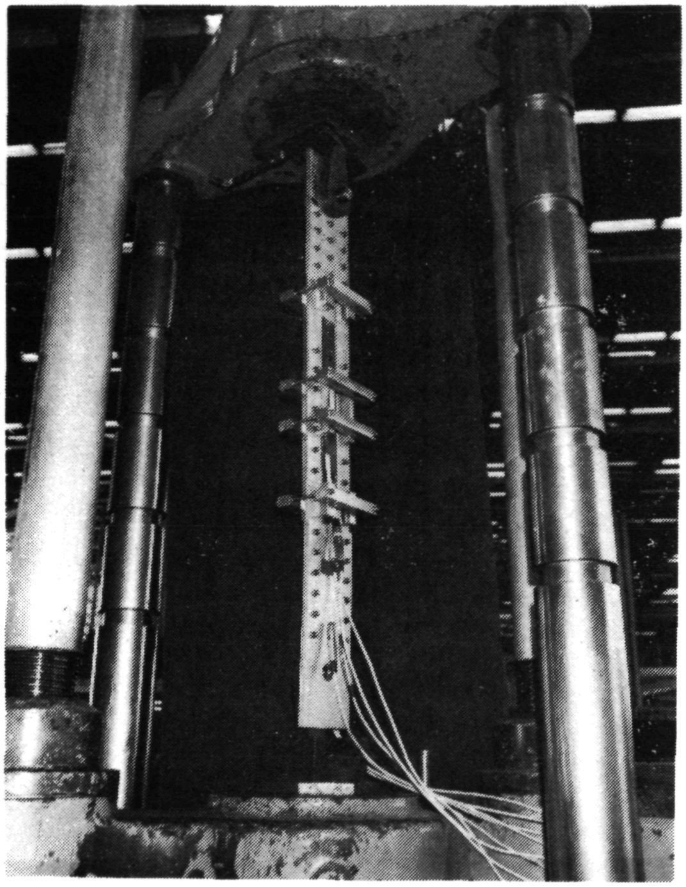


FIGURE 64. - GENERAL TEST ARRANGEMENT

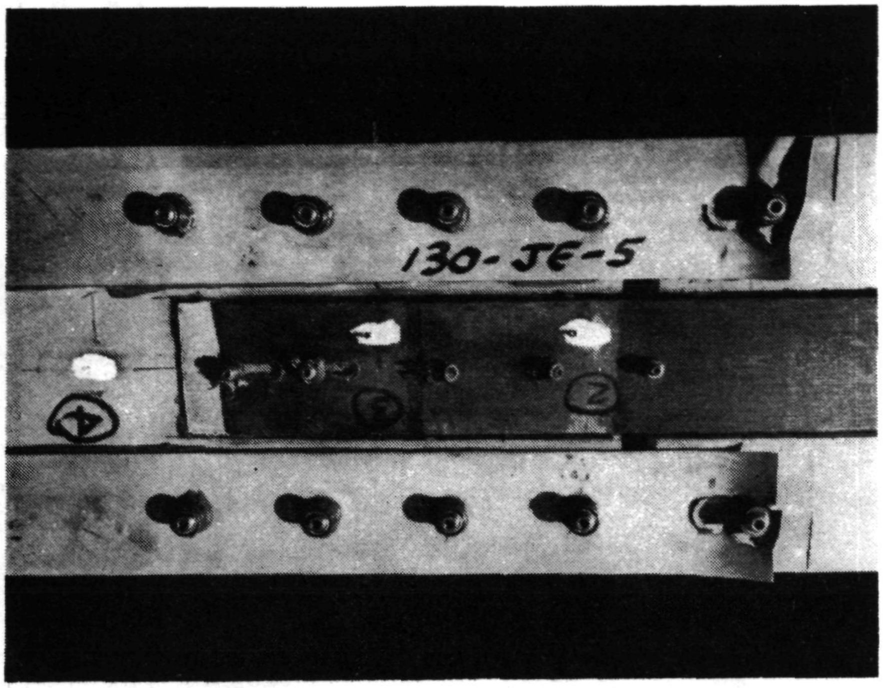


FIGURE 65. - 130-JE-5 FAILURE, LAMINATE SIDE

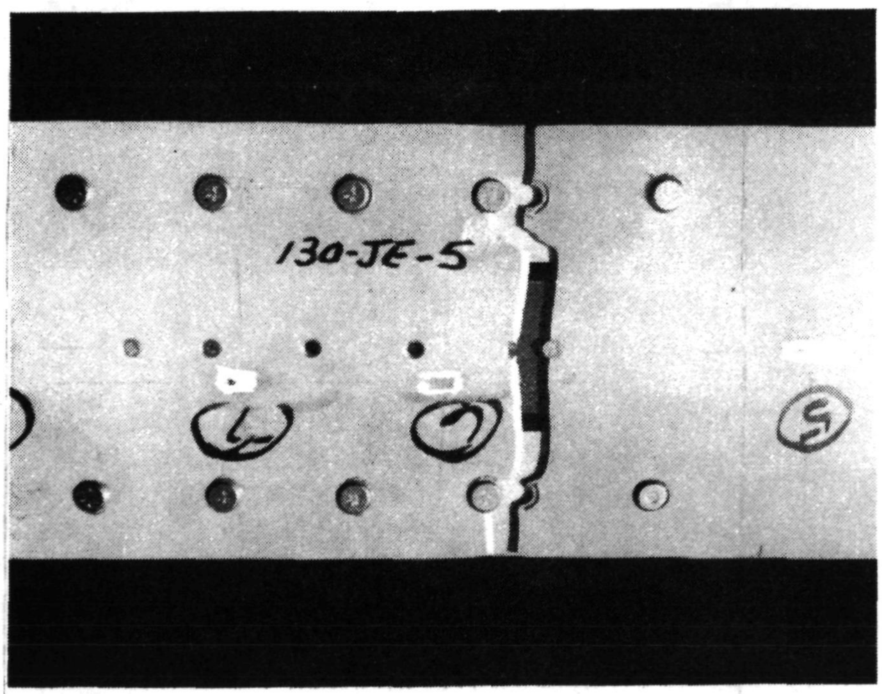


FIGURE 66. - 130-JE-5 FAILURE, SKIN SIDE

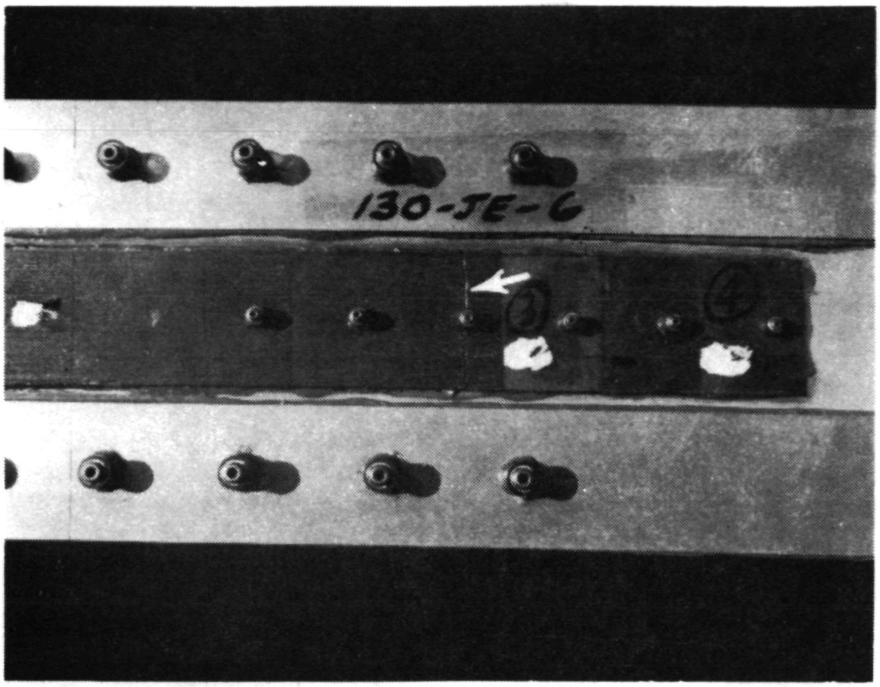


FIGURE 67. - 130-JE-6 FAILURE IN LAMINATE

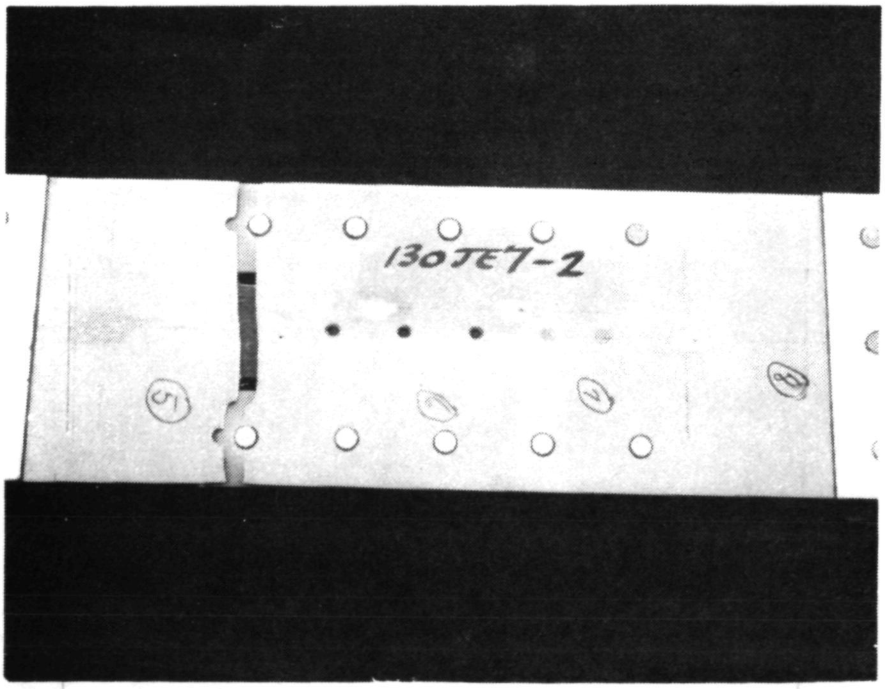


FIGURE 68. - SKIN SIDE OF FAILED 130-JE-7-2 SPECIMEN

As noted in the manufacturing section, some of the skin surfaces of the C-130 wing planks are severely contoured, and are not flat and parallel to the surfaces to which the laminates were to be bonded. During the bonding cycle, the ends of the panels were restrained. The resulting compressive stress in the aluminum causes lateral deflections in the thin areas of the skin planks if these areas are not adequately supported. Such deflections caused a number of disbonded areas in early bonded panels. Spot shimming in the thin areas was attempted to avoid this problem but was found to be only partially successful. Complete shim packages were then prepared and used on the outer skin surface of each plank. The addition of steel box channels on top of the panel to hold it flat to the tool eliminated this problem.

This difficulty resulted from the use of prototype tooling, and, in a full production program, when match-machined holding fixtures could be cost-effective, this problem would probably not have occurred. In future programs of this type, additional emphasis should be placed on full-size tooling concepts during the advanced development stage.

6.4 BONDLINE VOIDS & DISBONDS

No significant voids occurred in bonding the stringer laminates to stringer crowns. In the panel-to-laminate bonds, however, several unbonded areas occurred -- varying from minor edge voids to complete under-laminate unbonds. Due to the nature of the loading for this application, the edge voids were not in areas where they were critical from a structural standpoint. They were filled with EA 9309.1 room-temperature curing adhesive to minimize possible stress risers and used without further action.

The under-laminate voids were of two general types: small, enclosed areas, and larger areas where the unbond extends completely underneath the skin laminate. The small, enclosed voids were repaired by drilling and injecting with EA 9309.1. A single fastener was then installed through the repair to prevent void propagation. This repair is illustrated in Figure 69.

The more extensive voids were repaired as illustrated in Figure 70. Any old adhesive in the void areas was removed and replaced with EA 9309.1 room temperature curing epoxy adhesive. A 0.16 cm (0.063 in.) thick titanium doubler was bonded to the laminate over the void area, extending several inches beyond the void on either end. The entire repaired area was then fastened together with a double row of Hilok's spaced on about 3.2 cm (1.25 in.) centers.

As noted in the Manufacturing section, complex shimming of the contoured panels was used to provide suitable support during bonding. This support effectively eliminated the complete under-laminate voids and no disbonds of significance were found on either of the flight articles.

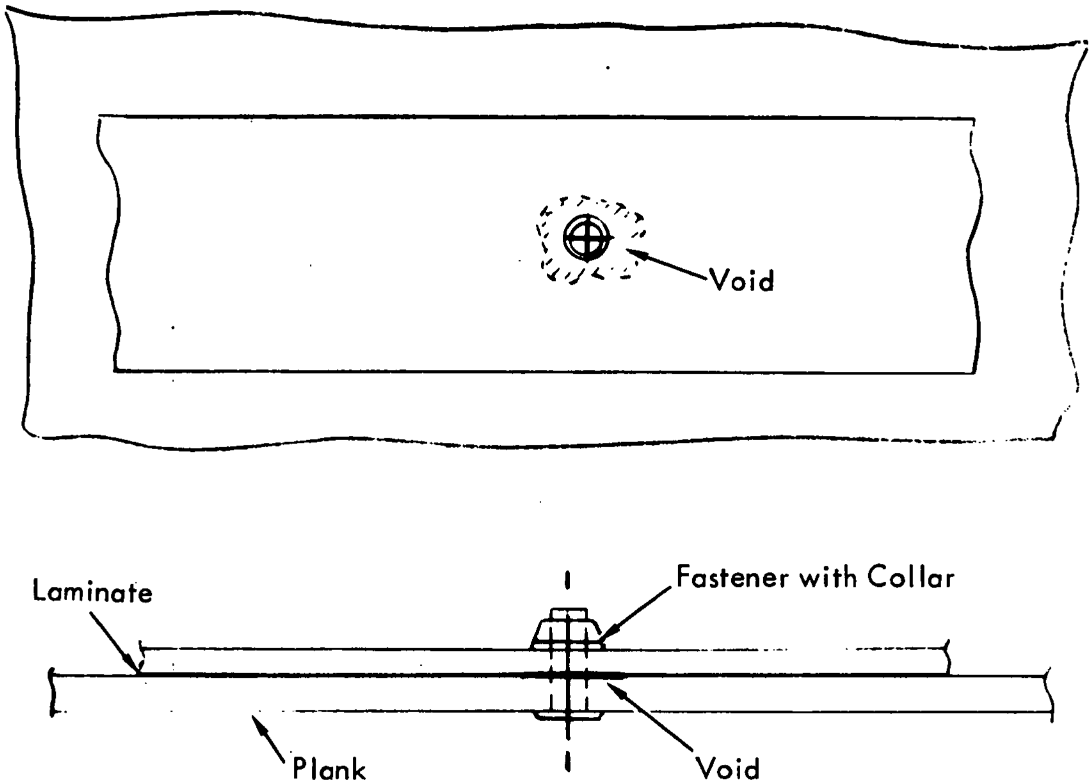


FIGURE 69. - REPAIR OF SMALL VOIDS

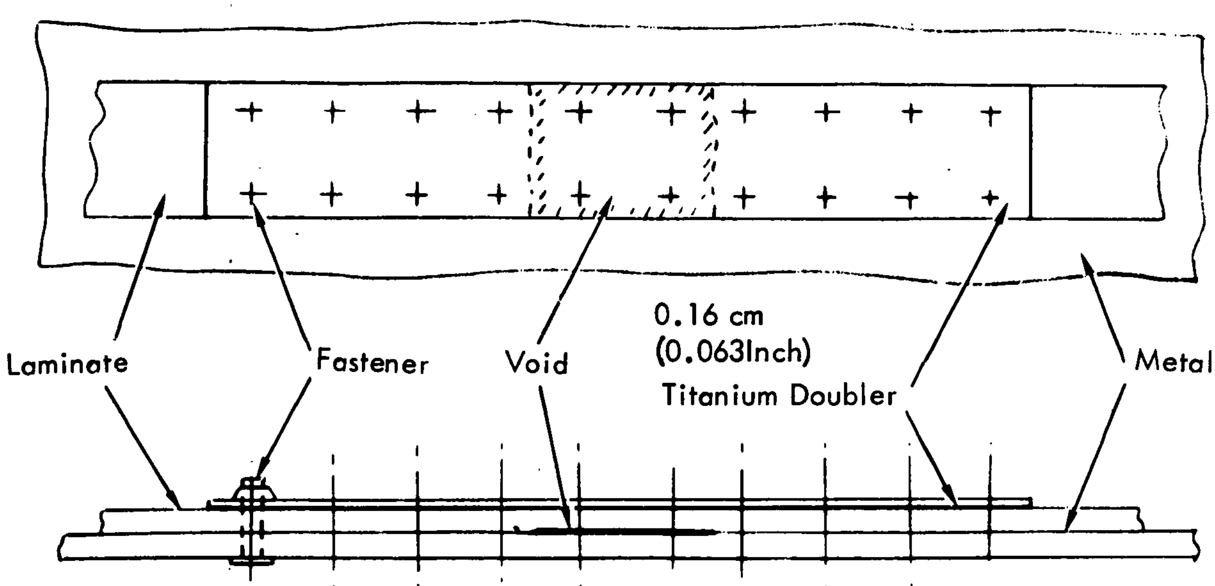


FIGURE 70. - REPAIR OF LARGER VOIDS

6.5 BORON-EPOXY TAPE

Some difficulty was encountered in working with material from the first shipment of 5.08 cm (2-inch) tape, due to the fact that the tape was not centered on its release film. Also, apparently due to uneven scrim tension, the tape was not laid straight on the backing paper. In some short lengths of 5.08 cm tape, gaps appeared between adjacent narrow tapes, and the total tape width exceeded the width of the laminate dams.

Initially, the tape was planned to be unrolled from the package and positioned simultaneously in the mold. Since this method relied upon guiding the release film for positioning the tape, the factors noted above caused the tape to be improperly positioned in the mold, and to ride up the sides of the mold dams.

The difficulties associated with uneven tension appeared to be confined to a single machine "load" during tape production. The centering problem was resolved by modifying the tape-laying machine to add a series of guide rolls which removed the tape from its release film, and then guided the tape (not the release film carrying the tape) into lay-up position.

The problem of short lengths of tape within a roll being slightly wider than the dams was improved by the supplier but was never completely resolved. The operator, in laying tape into the dam, frequently had to stop the machine and (using teflon spatulas) manually work a segment of the tape into the curing tool. This problem continued throughout the entire program.

7.0 PROGRAM SUPPORT ACTIVITIES

7.1 STRUCTURAL REPAIR MANUAL

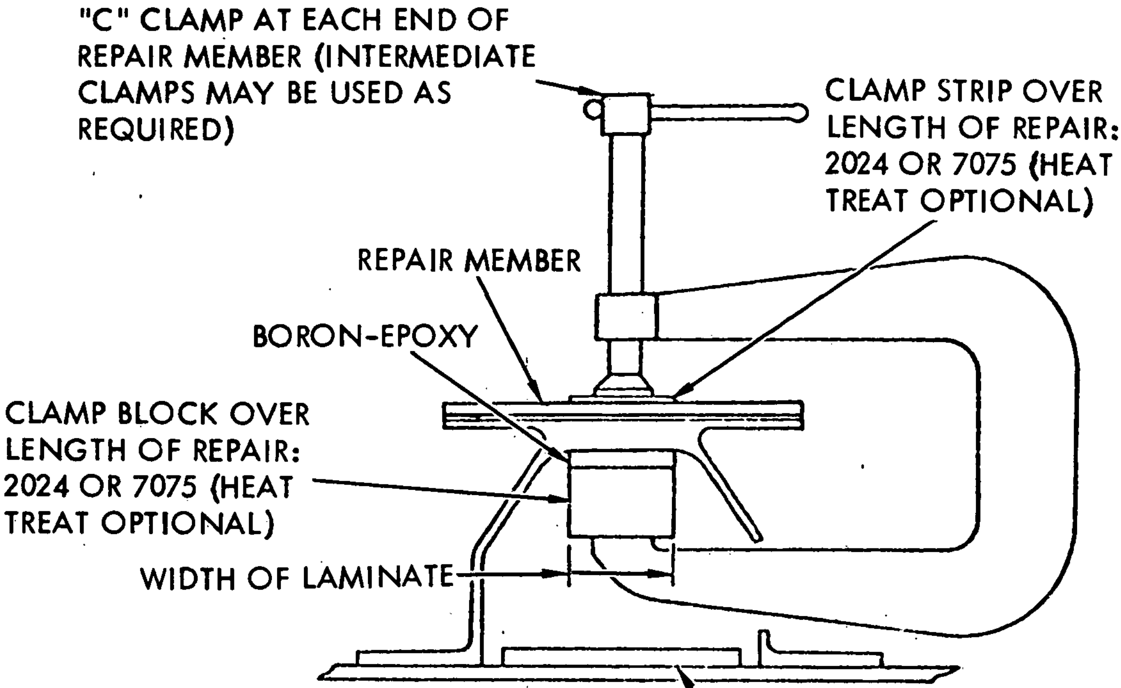
Standard repairs for the upper and lower surface skin panels and hat-section stringers (in areas where the boron-epoxy prohibited use of current standard repairs) were published in SMP 881, Reference 5. This document is to be used as a supplement to the aircraft Structural Repair Manual (T.O. 1C-130A-3) which covers areas not affected by the composite reinforcement.

In order to keep repairs as simple as possible, all repairs utilized standard metal repair techniques. No attempt was made to incorporate advanced composite technology into the field repairs. Certain precautions were noted to preclude the possibility of damaging the boron-epoxy laminate or its bond to the metal while making the repairs. Clamping techniques, illustrated in Figure 71, and drilling instructions were included for areas requiring that holes be drilled through the boron-epoxy. A typical skin-laminate-bond repair, extracted from SMP881, is illustrated in Figure 72. Stringer repairs used a similar procedure.

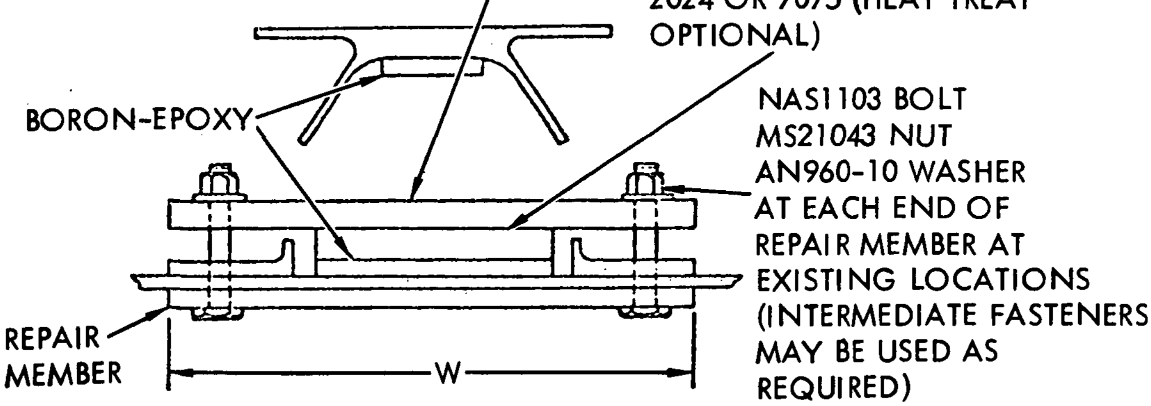
7.2 STRUCTURAL WEIGHT SAVING

Minor design changes to facilitate wing box production reduced the initially predicted weight saving of 229 Kg (506 lb) to 225 Kg (494 lb). Completed wing boxes were weighed to verify the predicted weight saving. In addition, the upper and lower surface assemblies were weighed separately for one wing box. The test article showed a weight saving of 222 Kg (488 lb). The surfaces of the two flight articles were approximately 17 Kg (38 lb) heavier than predicted, resulting in a weight saving of 208 Kg (456 lb).

Such slight weight variations are not unusual in large structural assemblies where scale accuracy and machining tolerance buildups can easily combine to produce much larger variations. On the center wing box, for example, a variation in the thickness of the skin panels of only +0.0127 cm (+0.005 in.) would cause a weight variance of 16 Kg (35 lb). The actual weights were, thus, well within manufacturing tolerances, and the predicted weights were satisfactorily demonstrated.

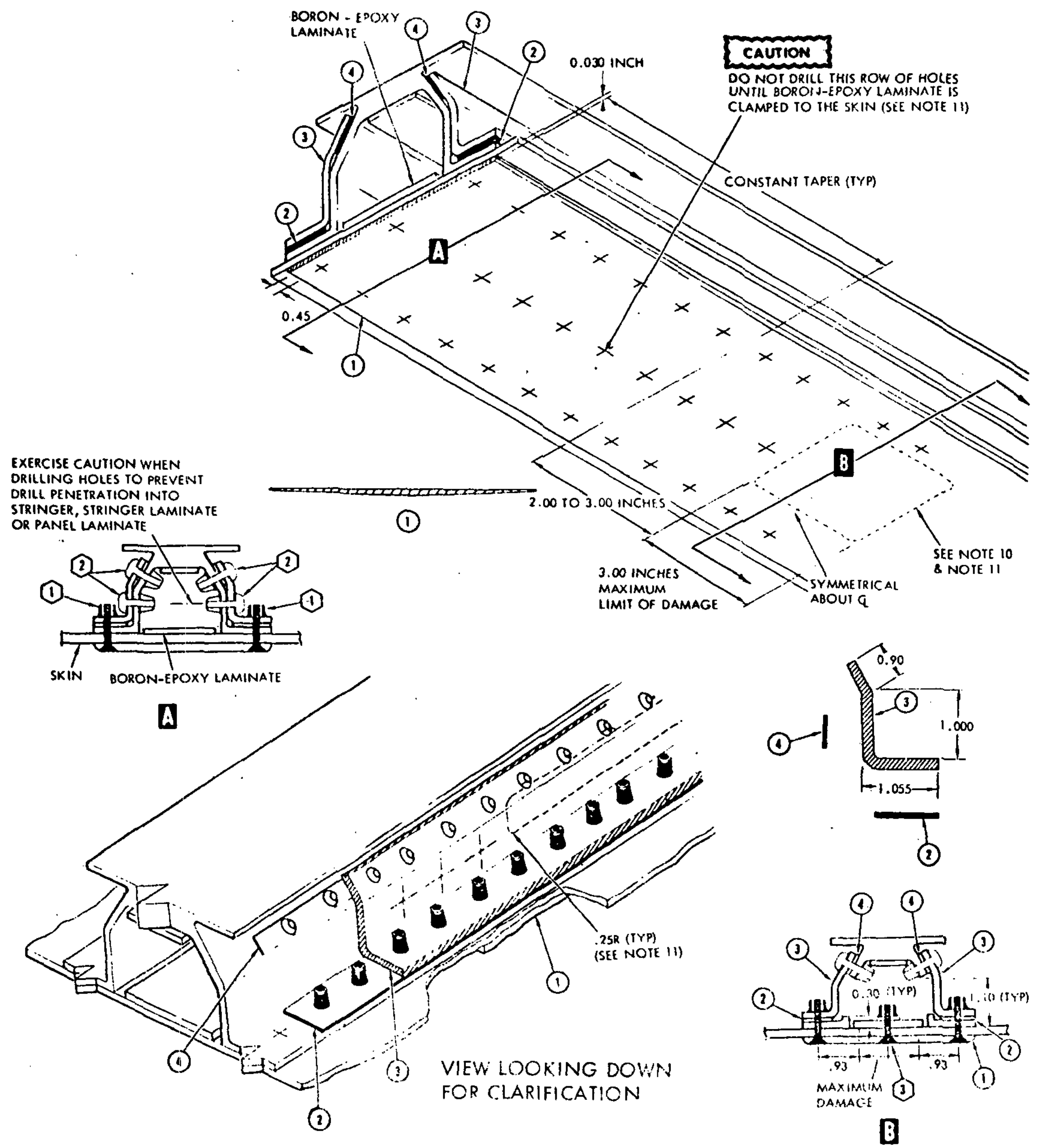


NOTE:
THESE CLAMPING METHODS ARE RECOMMENDED BUT DEVIATIONS MAY BE MADE THAT ACCOMPLISH THE SAME INTENT.



TYPICAL CLAMPING INSTRUCTIONS TO SUPPORT BORON-EPOXY PRIOR TO DRILLING

FIGURE 71.- SUGGESTED CLAMPING METHOD



NOTE

1. BEFORE INSTALLING REPAIR DOUBLER (1) FILL ALL COUNTERSUNK HOLES IN SKIN WITH EXPOXY RESIN MIL-R-9300. INSTALL REPAIR DOUBLER (1) WITH FLAT SIDE AGAINST THE SKIN.
2. EDGE SEAL REPAIR DOUBLER (1) WITH MIL-S-8802 FAYING SURFACE SEALANT.
3. NEW FASTENERS TO BE IN LINE WITH EXISTING FASTENER SPACING, EXCEPT AS NOTED.
4. THIS REPAIR IS NOT APPLICABLE AT SKIN SPLICE.
5. THIS REPAIR IS NOT APPLICABLE IF SKIN COMMON TO STRINGER LEG IS DAMAGED.
6. MAINTAIN A MINIMUM EDGE DISTANCE OF 1.5D TO 2D, WITH "D" EQUAL TO THE DIAMETER OF THE APPLICABLE FASTENER. EDGE DISTANCE IS MEASURED FROM ℄ OF HOLE TO EDGE OF PART.
7. FINISH REPAIR AREA IN ACCORDANCE WITH THE INSTRUCTIONS IN APPLICABLE SECTION OF GROUND HANDLING, SERVICING, AND AIRFRAME MAINTENANCE MANUAL.
8. WET INSTALL FASTENERS AND FAY SURFACE SEAL REPAIR MEMBERS PER T.O. 1C-130A-3, SECTION VII.
9. FOR CORROSION CONTROL, SEE T.O. 1C-130A-23.
10. DO NOT TRIM OUT DAMAGED AREA. STOP DRILL CRACKS WITH A NO. 30 (0.1285) DRILL.
11. STRINGER WALLS ARE CUT OUT TO PROVIDE THE ACCESS NECESSARY TO CLAMP THE BORON-EPOXY LAMINATE TO THE SKIN PRIOR TO DRILLING HOLES COMMON TO THE SKIN AND SKIN LAMINATE.

REPAIR MEMBERS

(1) REPAIR DOUBLER
THICKNESS EQUAL TO SKIN THICKNESS AT DAMAGED AREA   PLUG 0.020 X 3.10 X LENGTH
TO PICK UP 10 FASTENERS PER ROW EACH SIDE OF DAMAGED AREA. 7075-T73 SHEET QQ-A-250/13

(2) REPAIR STRAP
0.030 X 0.85 X LENGTH TO PICK UP 16 FASTENERS EACH SIDE OF DAMAGED AREA. 7075-T73 SHEET QQ-A-250/13

(3) REPAIR MEMBER
MAKE FROM LS5267-1 X LENGTH TO PICK UP 14 FASTENERS PER ROW EACH SIDE OF DAMAGED AREA. 7075-T73511 EXTRUSION QQ-A-200/11

(4) REPAIR STRAP
0.060 X 0.75 X LENGTH TO PICK UP 16 FASTENERS EACH SIDE OF DAMAGED AREA. 7075-T73 SHEET QQ-A-250/13

REPAIR FASTENERS

(1) HL911-6 HI-LOK PIN
HL70-6 COLLAR
WHERE EXISTING FASTENER IS TAPER-LOK USE: TL110-3 TAPER-LOK, KFN 542-3 NUT

(2) MS 90354-06 BLIND FASTENER

(3) HL911-6 HI-LOK PIN
HL70-6 COLLAR
AN960D10 WASHER UNDER COLLAR

(EXTRACTED FROM SMP-881: ALL DIMENSIONS ARE INCHES UNLESS OTHERWISE NOTED.)

FIGURE 72. - TYPICAL SKIN REPAIR IN AREA OF BORON-EPOXY LAMINATE

7.3 COST/PRODUCIBILITY DEVELOPMENT

7.3.1 Labor Cost Estimates

Manhours expended for the composite fabrication and assembly of the two flight articles were documented. This data is presented in Appendix D. The data presented is the average for the two flight articles. Summary data is shown in Table IX.

TABLE IX.-TOTAL FABRICATION TIME FOR COMPOSITE REINFORCEMENT OPERATIONS ON C-130 CENTER WING

	Part	Quantity Per Aircraft	Average Fabrication Man-Hours/Part	Total Fabrication Time (Hours)
Upper Surface	Fabricate Titanium Shims	52	.39	20.28
	Boron Laminates	26	22.52	585.52
	Bond Laminates to Hat Section Stringers	17	25.15	427.55
	Bond Laminates to Upper Surface Panels	4	92.70	370.80
Lower Surface	Fabricate Titanium Shims	102	.39	39.78
	Boron Laminates	38	22.52	855.76
	Bond Laminates to Hat Section Stringers	19	25.15	477.85
	Bond Laminates to Lower Surface Panels	3	92.70	278.10
			Total Fabrication Time =	3055.64

Note: Fabrication times are average figures for the two flight articles.

Cost estimates for production quantities of C-130E center wing boxes reinforced with boron-epoxy have been made based on the actual cost data accumulated during the manufacturing phase of two flight article wing boxes. These manufacturing manhours were projected on an 80% improvement curve for production quantities. To develop first unit cost it was assumed that the first production unit was the equivalent of the third unit produced. Basic C-130E aluminum recurring costs remain unchanged. Consequently, all costs shown are cost increments to the aluminum baseline.

Table X shows the distribution of manhours for each area of basic manufacturing operations. The manhours are for composite fabrication and assembly operations and are cumulative average for 200 center wing boxes.

Figure 73 shows the man-hours required to reinforce the C-130 center wing box with boron-epoxy for increasing quantities of production units. Table XI summarizes the manhours for specific quantities as shown.

TABLE X. - PROJECTED MANHOUR DISTRIBUTION FOR COMPOSITE FABRICATION AND ASSEMBLY OPERATIONS, CUMULATIVE AVERAGE FOR 200 AIRCRAFT

	Upper Surface		Lower Surface	
	Manhours	Percent	Manhours	Percent
Titanium Shim Fabrication	4	1	7	2
Boron-Epoxy Layup	96	24	141	30
Cure Laminate and Bond Panels	202	50	208	44
Produce Holes	98	25	115	24
Total	400	100	471	100

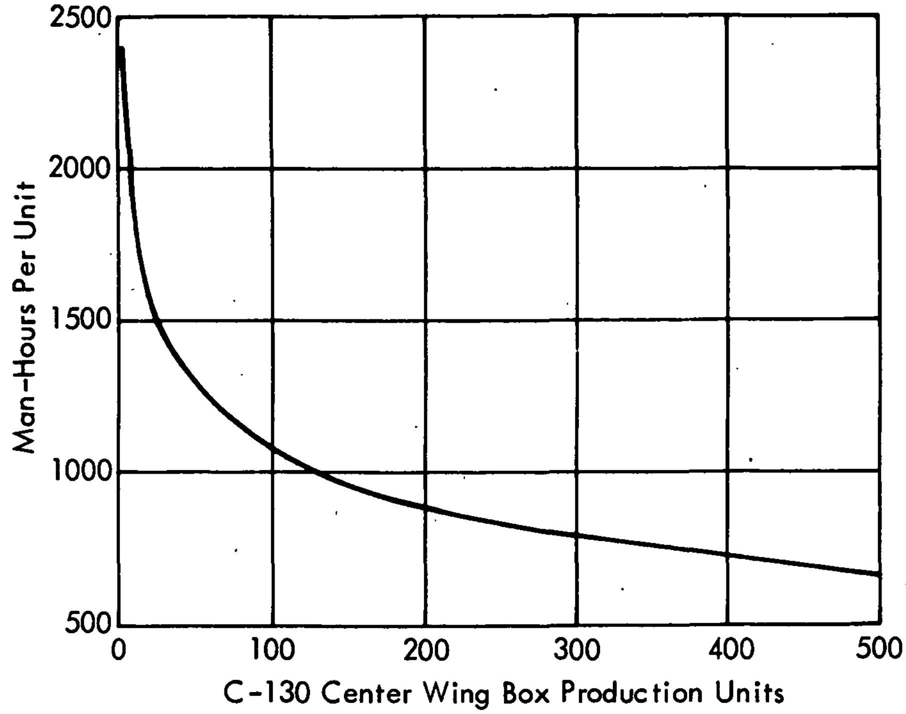


FIGURE 73. - COMPOSITE FABRICATION AND ASSEMBLY CUMULATIVE AVERAGE MAN-HOURS VERSUS C-130 CENTER WING BOX UNITS PRODUCED

TABLE XI. - SUMMARY OF FABRICATION MANHOURS

PRODUCTION QUANTITY	CUMULATIVE AVERAGE MANHOURS PER AIRCRAFT	UNIT MANHOURS PER AIRCRAFT
1	2384	2384
10	1842	1525
50	1282	952
100	1063	766
150	947	674
200	871	615

7.3.2 Material Cost Estimates

The C-130 center wing assembly used 159 kg (350 lb) of boron-epoxy preimpregnated tape; 85 kg (187 lb) in the upper surface and 74 kg (163 lb) in the lower surface. An actual material usage rate of 1.24, including scrap, was experienced on the two flight article wing boxes. A usage rate of less than 1.24 would be expected on a large quantity production program. However, using the material usage rate of 1.24 and an assumed cost for boron-epoxy tape of $50 per pound for 1975 production, a material cost of $21,700 per center wing results. Additional materials, such as adhesives, sealant, and titanium shim stock, add an estimated $1000 for a total material cost increase of $22,700 for a boron-epoxy reinforced center wing box.

7.3.3 Summary of Estimated Incremental Costs

The total cost increase to add boron-epoxy reinforcement to the C-130E center wing box is projected for the cumulative average for 200 units as follows:

Labor

871 manhours x $20/manhour = $17,420.

Material

$21,700 (boron-epoxy tape) + $1,000 (adhesive, etc.) = $22,700

Manufacturing Cost Increase (Cumulative Average for 200 Units)

Labor + Material = $40,120.

At a total weight saving of 229 kg (506 lb), the computed value per unit of weight saved is:

$40,120 ÷ 229 kilograms = $175/kilogram

$40,120 ÷ 506 pounds = $ 79.29/pound

REFERENCES

1. Harvill, W. E., et al., "Program for Establishing Long-Time Flight Service Performance of Composite Materials in the Center Wing Structure of C-130 Aircraft: Phase I - Advanced Development," NASA CR-112126, Lockheed-Georgia Company, November 1972.

2. Harvill, W. E., et al., "Program for Establishing Long-Time Flight Service Performance of Composite Materials in the Center Wing Structure of C-130 Aircraft: Phase II - Detailed Design," NASA CR-112272, Lockheed-Georgia Company, April 1973.

3. Petit, P. H., "An Application Study of Advanced Composite Materials to the C-130 Center Wing Box," NASA CR 66979, Lockheed-Georgia Company, June 1970.

4. Staff, Quarterly Progress Reports for NAS 1-11100 Contract, "Advanced Composite Reinforcement of C-130 Center Wing Box," Lockheed-Georgia Company:

   - First QPR: Phase I - Advanced Development, November 1971
   - Second QPR: Phase I - Advanced Development, February 1972
   - Third QPR: Phase II - Detailed Design, August 1972
   - Fourth QPR: Phase II - Detailed Design, November 1972
   - Fifth QPR: Phase II - Detailed Design, February 1973
   - Sixth QPR: Phase II - Detailed Design & Phase III - Fabrication, May 1973
   - Seventh QPR: Phase III - Fabrication & Phase IV - Ground/Flight Acceptance Tests, August 1973.
   - Eighth QPR: Phase III - Fabrication & Phase IV - Ground/Flight Acceptance Tests, November 1973.
   - Ninth QPR: Phase III - Fabrication & Phase IV - Ground/Flight Acceptance Tests, February 1974.
   - Tenth QPR: Phase III - Fabrication & Phase IV - Ground/Flight Acceptance Tests, May 1974.
   - Eleventh QPR: Phase III - Fabrication & Phase IV - Ground/Flight Acceptance Tests, August 1974.

5. Staff, SMP 881, "Structural Repair Instructions for Boron-Epoxy Reinforced Center Wing Skin Panels," Lockheed-Georgia Company, July 1974.

APPENDIX A

RELATIONSHIP BETWEEN SI UNITS AND U.S. CUSTOMARY UNITS

BASIC SI UNITS		
Physical Concept	Measurement	Abbreviation
Length	meter	m
Mass	kilogram	kg
Time	second	s
Force	Newton	N
Thermodynamic Temperature	degree Kelvin	^{o}K
Density	kilograms/meter^3	kg/m^3

PREFIXES		
Factor By Which Unit Is Multiplied	Prefix	Symbol
10^9	giga	G
10^6	mega	M
10^3	kilo	k
10^2	hecto	h
10	deca	da
10^{-1}	deci	d
10^{-2}	centi	c
10^{-3}	milli	m
10^{-6}	micro	μ

CONVERSION FACTORS		
To Convert From	To	Multiply By
Celsius (temp.)	kelvin	$t_K = t_c + 273.15$
Fahrenheit (temp.)	kelvin	$t_K = (5/9)(t_F + 459.67)$
foot	meter	3.048×10^{-1}
inch	meter	2.54×10^{-2}
pound mass (lbm avoirdupois)	kilogram	4.536×10^{-1}
pound mass force (lbf)	newton	4.44822
lbm/inch^3	kilogram/meter^3	2.768×10^4
psi	newton/meter^2	6.895×10^3

APPENDIX B

MANUFACTURING PROCESSES FOR BORON-EPOXY REINFORCED C-130 CENTER WINGS

This appendix outlines the step-by-step procedures followed in providing boron-epoxy reinforced center wing planks (panels) and stringers to the C-130 assembly line. The activities listed are, in general, those which are different from standard C-130 production practice, although some areas of similarity exist in fabricating metal details. Assembly of the reinforced panels and stringers into complete center wings follows standard shop practices and is not listed herein. Only major steps are included in these lists; many of the steps contain numerous operations. For example, the sulfuric acid anodize step encompasses some 31 separate operations.

B.1 METAL PARTS PROCESSING

B.1.1 Skin Panel Fabrication

1. Locate shop-aid hold-down tool on bed of numerically controlled machine tool.
2. Locate extrusion in hold-down tool.
3. Machine inside surface of part.
4. Relocate part with machined surface on bed of machine.
5. Machine taper on outside surface.
6. Machine lap joint on one edge.
7. Machine one edge net.
8. Machine opposite edge net.
9. Scribe each end of part.
10. Drill and ream tooling holes for doors.
11. Identify part number.
12. Rout to specified length.
13. Mill excess on edge at each side of doors.

14. Mill chamfers.

15. Blend mismatch and radii.

16. Deburr.

17. Identify electroetch part number.

18. Identify tag and lead seal.

19. Inspect - dimensional.

20. Vapor degrease and alkaline clean.

21. Inspect,

22. Prepare for penetrant inspection.

23. Inspect - penetrant.

24. Shot peen per blueprint.

25. Inspect.

26. Sulfuric acid anodize.

27. Identify - rubber stamp.

28. Inspect hardness.

29. Apply spray coat protective maskant.

30. Identify with metal tag.

31. Inspect.

32. Stock.

B.1.2 Stringer Fabrication

1. Saw - extrusion to specified length.

2. Identify - metal tag.

3. Layout machine cut areas.

4. Machine top of hat.

5. Machine chamfers.

6. Machine inside surface of stringer.

7. Deburr and break sharp edges.

8. Identify - metal tag.

9. Inspect - dimensional.

10. Mask for shot peen per blueprint.

11. Shot peen.

12. Demask.

13. Inspect.

14. Sulfuric acid anodize.

15. Identify - rubber stamp.

16. Inspect hardness.

17. Apply spray coat protective maskant.

18. Identify - metal tag.

19. Inspect.

20. Stock.

B.1.3 Titanium Shim Fabrication

1. Layout and fabricate per blueprint dimensions.

2. AFQA witness.

3. Inspect dimensionally.

4. Drill and ream holes.

5. Deburr.

6. Identify - electroetch.

7. Inspect.

8. AF QA witness.

9. Clean for metalbond. Record date and time of cleaning.

10. Inspect.

11. Apply adhesive primer. Record date and time applied and batch number of primer.

12. Inspect.

13. Wrap in clean Kraft paper.

14. Stock.

B.2 LAMINATE FABRICATION

B.2.1 Laminate Layup Sequence

1. Procure required primed titanium shims.

2. Apply adhesive to titanium shims.

3. Record type, batch number, roll number, and weight of adhesive.

4. Record time and date adhesive is applied.

5. Inspect.

6. Layup boron tape using layup machine and appropriate dam.

7. Cut tape lengths as shown by scribed marks on dams.

8. Install titanium shims in proper location.

9. Punch holes in tape plies (through holes in titanium doublers) using coordinated shop aid punch guide.

10. Place pin in layup to maintain doubler to ply location.

11. Continue layup operations 6 through 10 until all plies and doublers are in place.

12. Record boron-epoxy tape control number, batch number, and roll number.

13. Inspect (layup - hole location and quality),

14. AF QA witness.

15. Layup on autoclave platen.

16. Plug all open holes with teflon dowel.

17. Lay one ply of armalon (teflon coated fiber glass) on each of the laminate assemblies.

18. Place resin bleeder (one ply of 120 glass cloth for every ten plies of boron-epoxy tape on the layup) on the laminate assemblies in the dams.

19. Place 0.159 cm (0.0625 in.) thick layer of rubber in the dams the length of the assembly.

20. Place 0.318 cm (0.125 in.) thick aluminum caul plates over rubber the length of the dam.

21. Cover all assemblies with two layers of teflon to protect and make reusable next covering.

22. Cover all of assembly with four layers of 0.567 kg (20 ounce) glass cloth to cover sharp corners and to insulate parts.

23. Place a stainless steel chain around the periphery for air evacuation to vacuum ports.

24. Place a strip of vacuum bag sealing compound around the periphery of the tool.

25. Run thermocouple wires over sealing compound.

26. Place strips of bag sealing compound over thermocouple wires to achieve air seal.

27. Apply vacuum bag.

28. Apply vacuum and check for leaks.

29. Inspect.

B.2.2 Laminate Cure

1. Transport autoclave platen to autoclave.

2. Cure assembly in autoclave per following cycle.

   (a) Hold part under vacuum and check for leaks.

   (b) Maintain vacuum until autoclave pressure reaches 0.034 to 0.138 MN/m^2 (5 to 20 psi) - remove vacuum.

(c) Increase pressure to 0.586 MN/m^2 (85 psi) - hold during run within ±0.034 MN/m^2 (±5 psi).

(d) Start temperature cycle, increasing temperature at a rate of 2.78 to 5°K (5 to 9°F) per minute from 311 to 444°K (100 to 340°F).

(e) Hold temperature at 450 ± 5.56°K (350 ± 10°F) for 90 minutes.

(f) Cool to 339°K (150°F) holding 0.586 MN/m^2 (85 psi) pressure.

(g) Dump pressure - remove tool from autoclave.

3. Inspect bond cure cycle.
4. Record date, pressure, temperature, start, and completion time of cure and autoclave run number.
5. AF QA witness.
6. Debag.
7. Identify - rubber stamp.
8. Remove laminates from dams.
9. Remove plugs from holes in laminates.
10. Inspect - test specimens - record lab process request number.
11. Remove resin flash from laminates.
12. Ream holes.
13. Inspect holes.
14. Inspect - ultrasonic laminates.
15. AF QA witness.
16. Stock.

B.3 PREPARATION FOR BONDING

B.3.1 Preparing Metal Adherends

1. Layout bond area per blueprint - coordinate to machine cuts.
2. Inspect.
3. Strip spray coat maskant from bond areas.
4. Remove sulfuric acid anodize from bond areas.
5. Apply chromic acid anodize in bond areas.
6. Inspect.
7. Apply adhesive primer.
8. Record date primer applied and batch number of primer.
9. Remove balance of spray coat maskant.
10. Inspect.
11. Shim outer surface to flat condition (panels only).
12. Wrap in Kraft paper.
13. Stock.

B.3.2 Preparing Laminates

1. Remove peel ply from bond surface.
2. Cut adhesive film to proper width.
3. Apply adhesive film.
4. Record time, date of application of adhesive.
5. Record batch, weight, roll, and type of adhesive.
6. Inspect.

B.4 BONDING SEQUENCE

B.4.1 Bonding Laminates to Skin Panels

1. Position panel on cool tool with phenolic insulation strip at each end.
2. Unclamp and adjust end restraint rail to positive holding position and reclamp.
3. Remove protective peel ply from adhesive.
4. Position laminates on panel.
5. Tape laminates down with teflon tape strips on each edge of laminate.
6. Place spacers between laminates to maintain proper spacing relationship.
7. Position adhesive flow control metal strips adjacent to laminate edges and tape down with teflon tape.
8. Position pneumatic pressure hoses on top of laminates.
9. Place aluminum channels on top of pressure hoses and reinforce both channels and basic panels with steel box channel.
10. Place all laid-up test specimens in position on cool tool.
11. Place pneumatic pressure hose on top of test specimen bond areas.
12. Place aluminum channel on top of pressure hose.
13. Install (24) thermocouples and chart locations.
14. Position fiberglass tooling cloth insulation on top of panel, laminates, test panels, etc.
15. Place folded fiberglass insulation strips in position for steel clamp bars.
16. Place predetermined shim package on top of test specimen channel in location of clamp bars.
17. Install and torque clamp bars.
18. Install polyurethane insulation batts on top of fiberglass insulation and between clamp bars.
19. Make temperature monitoring recorder/thermocouple hookups.

20. Make power supply hookup to cool tool heater blankets.

21. Apply 0.241 MN/m^2 (35 psi) pressure to all hoses and check for leaks.

22. Apply and adjust current flow to heater blankets.

23. Start recorder.

24. Attain 386 ± 8.33°K (235 ± 15°F) and hold for 90 minutes under 0.241 MN/m^2 (35 psi).

25. Cut power at end of cure cycle and allow to cool to 339°K (150°F) before reducing pressure to ambient.

26. Disassembly setup which is the reverse of the assembly setup.

27. Remove panel assembly from cool tool and ultrasonic inspect for voids.

28. AF Q. A. witness.

29. Identify.

30. Inspect.

B.4.2 Bonding Laminates to Stringers

1. Place five full length stringer sections on cool tool with phenolic insulation strip at each end.

2. Adjust end restraint rail to positive holding position, shimming ends of stringers as required, and clamp.

3. Remove protective peel ply from adhesive on laminates.

4. Position laminates in stringers.

5. Tape laminates down with teflon tape to hold laminate in place.

6. Place test specimens in position on cool tool.

7. Place thermocouples at bond lines and chart locations.

8. Place pneumatic pressure hoses on top of laminates.

9. Place aluminum strips over pressure hoses.

10. Place pressure hose on top of test specimens.

11. Lay fiberglass tooling cloth insulation over total assembly.

12. Add folded glass insulation strips in areas where steel restraining bars are to be placed.

13. Place steel restraining bars over assembly, shimming where necessary.

14. Place polyurethane insulation batts on top of fiberglass insulation between steel restraining clamps.

15. Make temperature monitoring recorder/thermocouple hookups.

16. Apply 0.241 MN/m^2 (35 psi) pressure to all hoses and check for leaks.

17. Turn current on to heater blankets.

18. Heat parts, using bond line temperatures being shown on recorder, to $386 \pm 8.33^{\circ}K$ ($235 \pm 15^{\circ}F$) and cure for 90 minutes at 0.241 MN/m^2 (35 psi).

19. Cool to $331^{\circ}K$ ($150^{\circ}F$) after completion of cure cycle.

20. Disassemble.

21. Remove assemblies.

22. Ultrasonic inspect.

23. AF QA witness.

24. Identify.

25. Inspect.

B.5 FINISHING SEQUENCES

B.5.1 Installing Laminate End Fasteners

1. Place part on drill table.

2. Remove all teflon plugs from holes in laminate.

3. Locate shop aid drill block with drill rod.

4. Drill and ream per Blueprint.

5. Invert part.

6. Countersink all holes per B/P, if required.

7. Inspect.

8. Wet install fasteners per B/P.

9. Inspect.

B.5.2 Bonding Doublers for Blind Fasteners

1. Clean doublers for bonding.

2. Inspect.

3. Wrap in clean Kraft paper.

4. Prepare room temperature cure adhesive (EA 9309.1).

5. Apply adhesive to bond locations per B/P.

6. Position doubler and locate with loose rivet.

7. Remove excess adhesive and allow to cure.

8. Following cure, remove rivet.

9. Inspect.

B.5.3 Sealing Bondline Edges

1. Remove excess adhesive adjacent to laminate strip, if required.

2. Apply sealant to bond line per B/P.

3. Allow to cure.

4. Identify by rubber stamp.

5. Inspect.

B.5.4 Finish Machining Stringers

1. Cut to length, where required.

2. Machine one end per B/P.

3. Machine opposite end per B/P.

4. Inspect.

5. Touchup alodine machine cuts.

6. Identify with rubber stamp.

7. Inspect.

B.5.5 Finish Machining Panels

1. Set up panel on radial arm router.

2. Install router block.

3. Rout one end to B/P dimension.

4. Rout opposite end to B/P dimension.

5. Rout fuel access door cutouts where applicable.

6. Inspect.

B.5.6 Applying Finish Coating to Panels

1. Touchup alodine machined areas.

2. Hand clean.

3. Prime.

4. Paint per B/P.

5. Oven cure.

6. Identify with rubber stamp.

7. Inspect.

8. Stock.

B.5.7 Applying Finish Coating to Stringers

1. Hand clean (use 2-rag method)

2. Prime.

3. Paint.

4. Oven cure.

5. Identify with rubber stamp.

6. Inspect.

7. Stock.

APPENDIX C

PHASE III MRB ACTIONS

A summary of the MRB actions during the fabrication phase is shown in Table C-I. A detailed listing is also included which shows by category the number of DR's, the variation, and the disposition.

TABLE C-I.-PHASE III MRB ACTIONS SUMMARY

Category	1973	1974					
		Jan./Feb.	March	April	May	June	Total
Test Specimen, Tooling, and Materials	7	4	-	-	-	-	11
Laminates	3	25	3	-	-	-	31
Bonding	3	24	11	2	-	-	40
Metal Parts/Processing/Assembly	11	15	23	24	14	5	92
Total	24	68	37	26	14	5	174

Test Specimen, Tooling, and Materials: 11 Total

No. of DR's	Variation	Disposition
7	ti-to-ti lap shear test specimen failed below spec.	Use based on additional test specimen results
1	One specimen in one laminate run - low flexural value	Use based on high average value
1	2 short laminate assemblies - tool error	Scrap - correct tool
2	ti-to-ti test panel omitted in one laminate run	Use based on results from other test specimen

Laminates: 31 Total

No. of DR's	Variation	Disposition
5	Fastener holes in laminates and/or ti shims out of tolerance	Scrap
17	Fastener holes in laminates and/or ti shims out of tolerance	Repair and Use
6	Fastener holes in laminates and/or ti shims out of tolerance	Use as is
1	Frayed and delaminated edge in laminate	Repair and Use
1	Damaged laminate edge	Repair and Use
1	Temperature out of tolerance during 1 laminate run	Use based on PC results

Bonding: 40 Total

No. of DR's	Variation	Disposition
2	Stringers damaged during bond cycle	Scrap
1	Laminate mislocated during bond cycle	Remove laminate, Repair, Use
7	Stringer ends damaged during bond cycle	Repair, Use
1	Doubler mislocated	Use
1	Lap shear specimen failed in one bond cycle due to lack of pressure - panels were pressurized	Use based on ultra-sonics and PC results
4	Temperature out of tolerance during bond cycle	Use based on PC results
2	Low spots	Install titanium over low spots - Use

Bonding (continued)

No. of DR's	Variation	Disposition
1	Ends of panels bent during bond cycle	Repair, Use
10	Voids between panel and laminate	Repair, Use
1	Laminate on panel binds on stringer assembly	Allow stringer to float off location - Use
1	Panel shifted during bond cycle	Remove and replace laminate - Use
7	Excessive adhesive squeeze out during bond cycle	Remove excess, Repair, Use
1	Fasteners on panel inaccessible	Repair, Use
1	Reinforcing hole blocks on stringer mislocated	Repair, Use

Metal Parts/Processing/Assembly: 92 Total

No. of DR's	Variation	Disposition
1	Stringer extrusion received with wrong material	Return to vendor
1	Titanium doublers wrapped in wrong paper	Reprocess and use with correct paper
3	Panels with thin skin thickness	Scrap
2	Stringer with gaps under Hilok heads	Use
1	Panels, machining set-up error	Scrap
4	Panels, machining set-up error	Use
4	Panels, machining set-up error	Repair, Use
1	Stringer, crown thickness undersize	Use

Metal Parts/Processing/Assembly (continued)

No. of DR's	Variation	Disposition
1	Stringer, flange bent	Repair, Use
4	Stringer, light to severe anodic burns	Repair, Use
1	Stringer, chemical burns - thin maskant	Use
2	Panel, 1 skin gage undersize, 1 undercut	Scrap
1	Stringer, crack in end of hat section	Repair, Use
4	Panels, minor gouges from router when removing excess adhesive	Repair, Use
1	Stringer assembly built up in reverse	Repair, Use
2	Panel, milled undersize	Use, Correct Type
1	Panel, holes oversize	Repair, Use
58	Miscellaneous assembly deviations including such items as short edge distances for fasteners, subcomponent interferences, use of substitute fasteners, etc.	Repair, Use

APPENDIX D

BASIC DATA - ACTUAL AVERAGE MANHOURS FOR COMPOSITE-REINFORCEMENT OPERATIONS

This appendix summarizes in tabular form the average manhours per part for reinforcing the aluminum stringers and wing planks with boron-epoxy. The manhours shown are average values for the two flight articles which were produced. The operations shown are those which are peculiar to the composite reinforcement process.

Part	Operation	Avg. Time/Part
Titanium Shims	Shear Shim (based on fab of 36 shims)	0.12 Hrs
	Drill and Ream	0.12
	Deburr	0.02
	Identify Electroetch	0.02
	Clean for Metal Bond	0.05
	Apply Adhesive Primer	0.05
	Protective Wrap	0.01
		0.39 Hrs
Boron Laminate Assembly	Apply Adhesive to Titanium Shims	0.50 Hrs
	Layup Boron-Epoxy Tape, Titanium Shims, & Process Control Specimens	12.50
	Punch Holes in Boron-Epoxy Tape	4.00
	Transfer Dam to Platen	0.12
	Apply Vacuum Bag & Check for Leaks	1.50
	Transport to Autoclave & Make Hookups	0.75
	Cure Laminate in Autoclave	0.75
	Return to Clean Room & Debag	0.50
	Unload Dam & Clean Laminate	0.50
	Cleanup Dam & Prepare for Reuse	0.80
	Remove Plugs & Ream Holes	0.40
	Identify with Stamp	0.20
		22.52 Hrs

Part	Operation	Avg. Time/Part
Wing Panel-to-Laminate Bond	Strip Protective Coating from Bond Area	5.25 Hrs
	Process for Metal Bond	2.00
	Apply Primer to Bond Area	.90
	Strip Remainder of Protective Coating	5.80
	Square Ends to Fit Cool Tool & Install Doubler at Panel Ends if Required	2.75
	Install Shim Package on Panel Skin Side	6.00
	Install Panel on Cool Tool & Adjust End Restraints	.80
	Apply Adhesive to Laminate	9.60
	Prepare and Position Process Control Test Panels	3.00
	Install Laminate on Panel & Tape Down	1.60
	Tape Down Adhesive Flow Control Strips	2.25
	Install Pressure Hose & Retainer Channel	1.75
	Install Thermocouples & Chart Locations	1.00
	Cover with Fiberglass Insulation Blanket	.90
	Install & Torque Clamp Bars	6.00
	Make Air & Electrical Hookups	.80
	Install Polyurethane Insulation Batts	.60
	Bond Cycle	12.40
	Disassemble Cool Tool Setup	8.20
	Remove Adhesive Tape & Metal Strips	1.20
	Drill, Ream & Install Fasteners	19.50
	Identify with Stamp	.40
		92.70 Hrs

Part	Operation	Avg. Time/Part
Stringer-to-Laminate Bond	Strip Protective Coating from Bond Area	1.60 Hrs
	Process for Metal Bond	.80
	Apply Primer to Bond Area	.45
	Strip Remainder of Protective Coating	1.25
	Apply Adhesive to Laminate	2.15
	Locate Stringer on Cool Tool	.40
	Place Laminate in Stringer & Tape Down	.40
	Prepare & Position Process Control Test Panels	.60
	Install Thermocouples & Chart Locations	.20
	Install Pressure Hose & Retainer Plate	.25
	Cover with Fiberglass Insulation Blanket	.20
	Install & Torque Clamp Bars	1.15
	Make Air & Electrical Hookups	.20
	Install Polyurethane Insulation Batts	.10
	Bond Cycle	2.40
	Disassemble Cool Tool Set-up	1.20
	Remove Adhesive Tape	.20
	Identify with Stamp	.20
	Machine End Cuts on Stringers	2.80
	Remove Plugs, Drill, and Ream	8.60
		25.15 Hrs